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CONSOLIDATED VULTEE

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MODEL

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TITLE

# NOTES ON LONG RANGE MISSILE DESIGN

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## INTRODUCTION AND CONTENTS

The present report is intended as a supplement to CVAC Report No. 48-35003 entitled: "Proposed Development Program on Rocket Type Missiles".

Some of the data of the above report are amplified or substantiated. In addition some further data are submitted underlying Convair's belief that the rocket engine is the power plant best suited for long-range bombardment missiles.

The topics discussed are:

# ROCKETS VS RAMJETS

## A COMPARISON BASED ON RAND STUDIES

## ROCKETS VS RAMJETS

## A COMPARISON BASED ON RAND STUDIES

~~As a supplement to previous comparisons of Rockets with other type missiles submitted in CVAC Report ZP-48-35003, ZP-6002-002, and "Convair Rocket Missiles",~~ A weight and economic comparison of Rockets vs Ramjets is presented herein based entirely on studies conducted under Project Rand.

It was assumed that identical propellants (Hydrazine-Oxygen) were used in the rocket missile and in the ramjet booster. Any future improvement in rocket propellants would be reflected equally in both types of missile.

Rather than using the present day diffusers the ramjet weights were based on the use of the "reflected wave" type diffuser, which according to RAND is "believed to be attainable in about five years". (Report RA-15072 page 21)

The gross weights vs range plotted in Figure 1 were taken from RAND plots: Figure 32A of Report RA 15063 for the rockets and Figure 211 of Report RA 15070 for the ramjets.

In addition to the weight comparison a cost comparison was made for missiles of 3,000 and 5,000 miles range. The results are shown in Figures 2 and 3.

The costs per pound used in these comparisons for airframe, power plant and guidance are based on CVAC experience to date in the guided missile field. The propellant costs represent current prices in carload lots with the exception of Hydrazine which is based on Project Rand estimates.

For lack of specific data the cost per pound of the ramjet power plant was assumed the same as that for airframe, even though several of the power plant components - such as valves, pumps, injectors - are of the same degree of complication as those of the rocket power plant.

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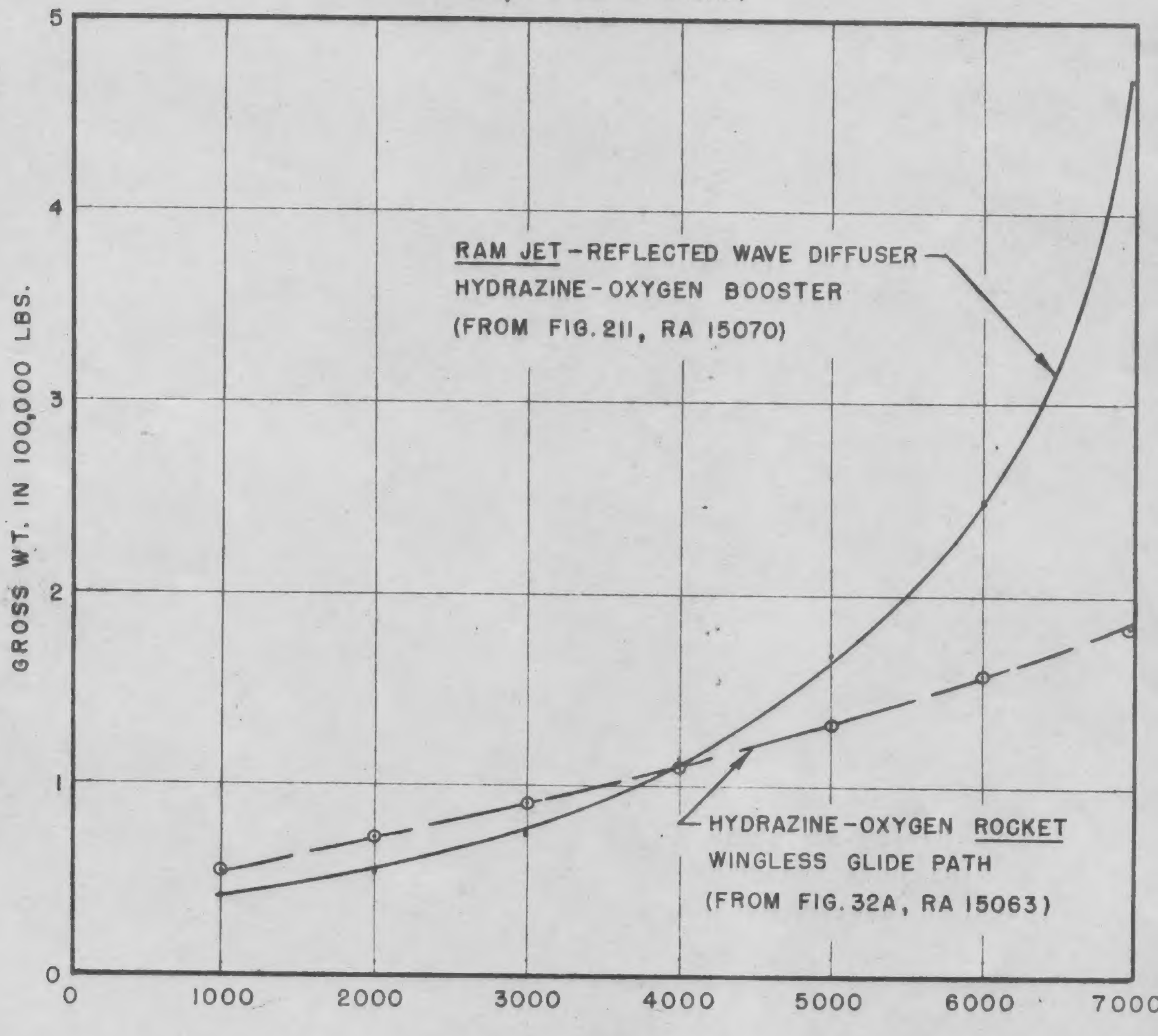
GROSS WEIGHT COMPARISON OF
ROCKETS AND RAM JETS
(10,000 LB. WARHEAD)



## 5,000 MILE ROCKET-RAMJET ECONOMIC COMPARISON

2-STAGE HYDRAZINE-OXYGEN ROCKET VS REFLECTED WAVE RAMJET WITH HYDRAZINE-OXYGEN BOOSTER
5,000 MILE RANGE - 10,000 POUND WARHEAD

| | Unit Cost | Ramjet Missile | | Booster | | Ramjet+Booster | | Rocket Missile | |
|---|---|---|---|---|---|---|---|---|---|
| | $/lb | Weight | Cost | Weight | Cost | Weight | Cost | Weight | Cost |
| Airframe | $ 27.00 | 11,609 | $314,000 | 9,560 | $258,000 | 21,169 | $ 572,000 | 12,892 | $348,000 |
| Rocket Power Plant | 80.00 | ---- | ---- | 6,010 | 481,000 | 6,010 | 481,000 | 6,400 | 512,000 |
| Ramjet Power Plant | 27.00 | 3,460 | 93,400 | ---- | ---- | 3,460 | 93,400 | ---- | ---- |
| Guidance | 100.00 | 850 | 85,000 | ---- | ---- | 850 | 85,000 | 850 | 85,000 |
| Payload | ---- | 10,000 | ---- | ---- | ---- | 10,000 | ---- | 10,000 | ---- |
| Total Wt. Empty | ---- | 25,919 | $492,400 | 15,570 | $739,000 | 41,489 | $1,231,400 | 30,142 | $945,000 |
| Oxygen | .04 | ---- | ---- | 40,300 | $ 1,600 | 40,300 | $ 1,600 | 54,858 | $ 2,190 |
| Hydrazine | .45 | ---- | ---- | 44,130 | 19,900 | 44,130 | 19,900 | 60,000 | 27,000 |
| Gasoline | .025 | 39,081 | $ 980 | ---- | ---- | 39,081 | 980 | ---- | ---- |
| Total Fuel | ---- | 39,081 | $ 980 | 84,430 | $ 21,500 | 123,511 | $ 22,480 | 114,858 | $ 29,190 |
| TOTAL | ---- | 65,000 | $493,380 | 100,000 | $760,500 | 165,000 | $1,253,880 | 145,000 | $974,190 |

FIGURE 2

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# FIGURE 3

## 3000 MILE ROCKET-RAMJET ECONOMIC COMPARISON

2-Stage Hydrazine-Oxygen Rocket vs. Reflected Wave
Ramjet with Hydrazine-Oxygen Booster

3000 Mile Range - 10,000 Pound Warhead

| | UNIT COST | RAMJET MISSILE | | BOOSTER | | RAMJET + BOOSTER | | ROCKET | |
|---|---|---|---|---|---|---|---|---|---|
| | $/Lb | Wt. | Cost | Wt. | Cost | Wt. | Cost | Wt. | Cost |
| Airframe | $ 27.00 | 4445 | $120,000 | 4400 | $119,000 | 8845 | $239,000 | 7160 | $193,000 |
| Rocket Power Plant | 80.00 | --- | --- | 3620 | 289,000 | 3620 | 289,000 | 5218 | 418,000 |
| Ramjet Power Plant | 27.00 | 1605 | 43,400 | --- | --- | 1605 | 43,400 | --- | --- |
| Guidance | 100.00 | 850 | 85,000 | --- | --- | 850 | 85,000 | 850 | 85,000 |
| Payload | --- | 10000 | --- | --- | --- | 10000 | --- | 10000 | --- |
| Total Wt Empty | --- | 16900 | $248,400 | 8020 | $408,000 | 24920 | $656,400 | 23228 | $696,000 |
| Oxygen | .04 | --- | --- | 19000 | $ 760 | 19000 | $ 760 | 32400 | $ 1,300 |
| Hydrazine | .45 | --- | --- | 20980 | 9,450 | 20980 | 9,450 | 35672 | 16,100 |
| Gasoline | .025 | 13100 | $ 327 | --- | --- | 13100 | $ 327 | --- | --- |
| Total Fuel | | 13100 | $ 327 | 39980 | $ 10,210 | 53080 | $110,537 | 68072 | $ 17,400 |
| Total | | 30000 | $248,727 | 48000 | $418,210 | 78000 | $666,937 | 91300 | $713,400 |

FIGURE 3

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## CONCLUSION:

The above comparisons indicate that for ranges up to 4000 miles the Hydraxine-Oxygen Rocket and the ramjet with "Reflected Wave" diffuser and Hydrazine-Oxygen booster are closely comparable as to gross weights. Above 4000 miles the ramjet weight rapidly becomes excessive.

The economic comparison for 5000 mile range missiles indicates the cost of the ramjet missile to be 29 percent greater than the cost of the rocket missile. At a 3000 mile range the rocket missile cost exceeds the ramjet cost by 7 percent. Judging from the shape of the curves in Figure 1 this differential should be approximately the same at 2000 mile range.

It is interesting to compare the RAND values with values attainable in the not so distant future. The CVAC glide rocket described in this report and the MX-770 ramjet were selected to this end. Inasmuch as these missiles have different ranges and different payloads they were compared on the basis of cost per ton mile of payload.

| | MISSILE | REFERENCE | COST IN $/TON MILE |
|---|---|---|---|
| GLIDE ROCKET | RAND 5000 mile | page 4 | 39.0 |
| | RAND 3000 mile | page 5 | 47.5 |
| | CVAC 2000 mile | page 45 | 54.5 |
| RAMJET | RAND 5000 mile | page 4 | 50.1 |
| | RAND 3000 mile | page 5 | 44.4 |
| | MX-770 1000 mile | ZP-48-35003 p. 36 | 261.3 |

These figures emphasize the fact that today the rockets are much closer than the ramjets to the ultimate perfection envisioned by RAND.

# COMMENTS ON THE STATE OF DEVELOPMENT OF RAMJETS FOR LONG RANGE MISSILES

## I. INTRODUCTION:

The selection of rocket propelled missiles over Ramjets for long range ground-to-ground application was based on considered study of the two methods of propulsion. As one of the principal Contractors in the Navy Bureau of Ordnance's Project Bumblebee, with responsibilities for missile fabrication and flight test, laboratory operation, and combustion research, Convair is peculiarly qualified to discuss the question of Ramjets. Coupled with two years of ramjet testing experience both at Convair operated Ordnance Aerophysics Laboratory at Daingerfield and the San Diego facility, Convair is fortunate in having developed and operated rocket power plants as part of MX-813 and MX-774 programs. This combined experience is valuable in selecting the most suitable propulsion method for long-range ground-to-ground missiles.

Convair's choice of rockets for the long-range bombardment type missiles, in addition to other considerations, is based on sound knowledge of the status of development and the possibilities of various propulsion methods. A brief review of the ramjet art is presented to review the thinking that went into this selection. However, it must be emphasized that the conclusions drawn are based on the requirements for long-range bombardment missiles only, and are not necessarily true for all ramjet applications. Ramjets can be shown to be the optimum power plants for other type missiles.

## II. SUMMARY AND CONCLUSION:

The following comments are intended to show that ramjets in their present stage of development are unsuitable for long range propulsion, and that many of the problems which have been indicated by the Bumblebee activity will remain serious obstacles even after a solution for the anti-aircraft ramjet weapon has been reached.

1. Flight tests have confirmed the laboratory findings most satisfactorily. They seem to indicate that present articles will not operate in their design Mach number range at altitudes significantly higher than 35,000 feet. 40,000

2. Experience with the scaling problem clearly indicates the need for additional knowledge before extrapolations to arbitrary size are possible. ~~Therefore, test facilities not yet under construction are a prerequisite to the development of large long-range ramjet missiles.~~

3. A failure of certain types of ramjet combustors to achieve ignition at launching Mach numbers has been discovered. Solutions to this problem may place some limitations on the launching configuration of some missiles.

4. While methods are under development for cooling the hot elements of the long-range ramjet, experience has shown that some loss in efficiency will almost certainly be sustained to accomplish such cooling.

5. The problem of combustion control has proved to be considerable, and the necessity is seen of always having to carry air mass flow sensing equipment and associated fine metering devices.

Reliability of all the generalized studies of ramjet range made to date is thus brought into cuestion. The refinements in design necessary to solve the foregoing problems and the flight limitations so far experienced make it unlikely that current paper solutions will be fulfilled in all respects and that the time scale for the development of useful weapons from the standpoint of propulsion alone may be many years.

## III. DEVELOPMENT STATUS:

The following review of the present state of development of ramjets gives a measure of the difficulties that will have to be surmounted to arrive at a suitable design for long-range ramjets.

A. Laboratory Tests: Observations during laboratory tests of propulsion units have lead to criteria of burner performance based upon minimum flight requirements of thrust and maneuverability, and what appear at the present laboratory stage to be practicable minima of burner performance. This laboratory experience in balancing combustion efficiency, air-fuel ratio range, practicability of manufacturing component parts, cost and time of development and general practical limitations has lead to a definition of a satisfactory burner as one which will attain a combustion efficiency greater than 75 percent over an air/fuel range of 2/1, e.g., from a/f = 30/1 to a/f = 15/1. A typical burner performance curve is presented in Figure 4. It is further required that it burn readily available fuel and meet severe starting conditions. This standard of efficiency and burning range while satisfactory for short-range ground-to-air missiles is by no means adequate for a long range missile where combustion efficiency should be around 90 percent.

This laboratory experience has generally been limited to ramjet engines with a tailpipe exit diameter of twenty-four inches or less at burner pressures greater than one atmosphere, while long-range ramjets will certainly be larger (it is generally estimated that they will be from four to six feet in diameter) and also must operate with burner pressures less than sea level atmosphere.

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Figure 4. Combustion Efficiency vs Equivalence Ratio For a Typical Laboratory Burner. Six - inch scale Designed For $M_o$ = 1.6, Maximum Thrust at Stoichiometric. (Maximum Thrust For This Burner Occurs at Equivalence Ratio 1.2)

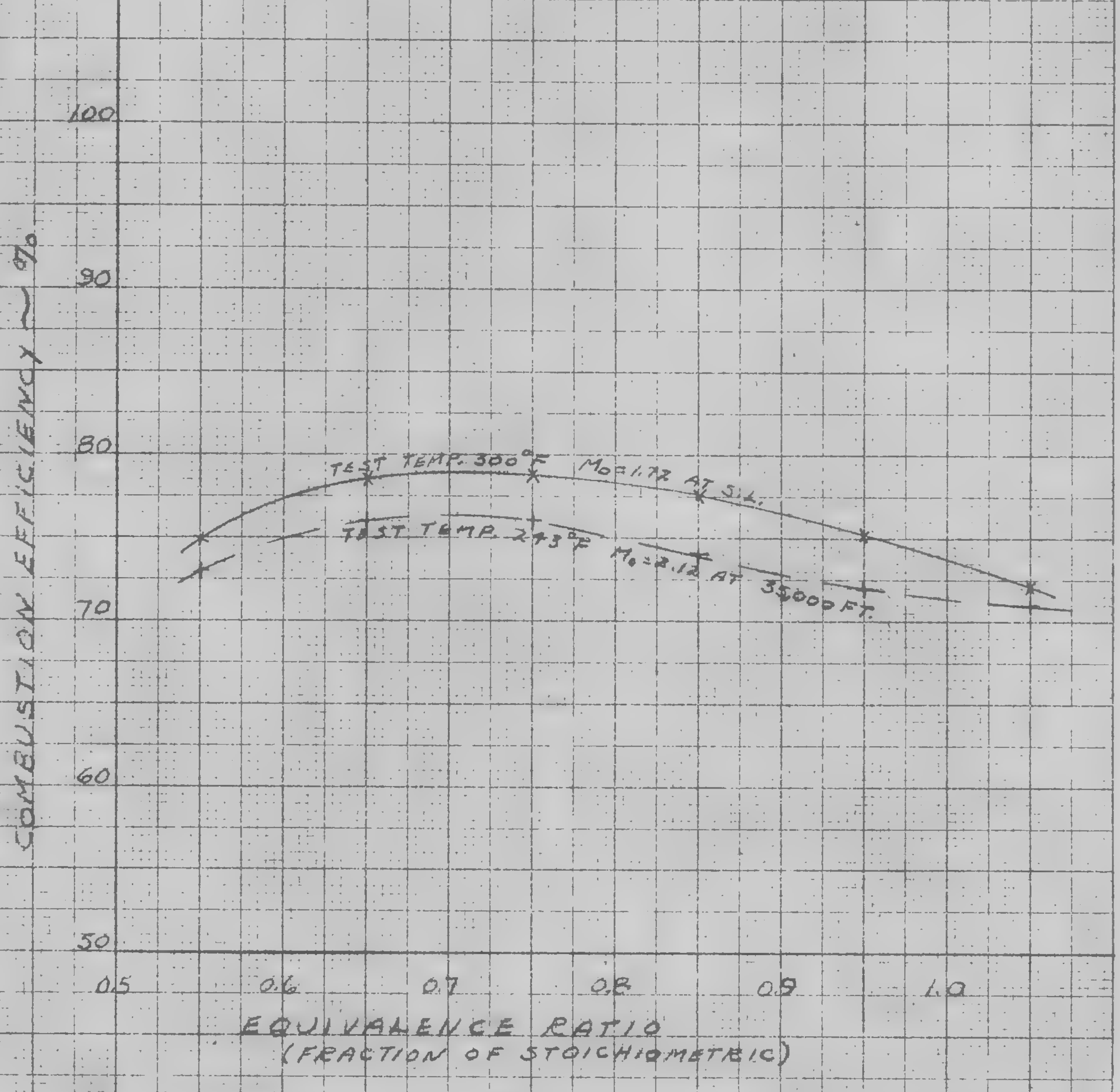


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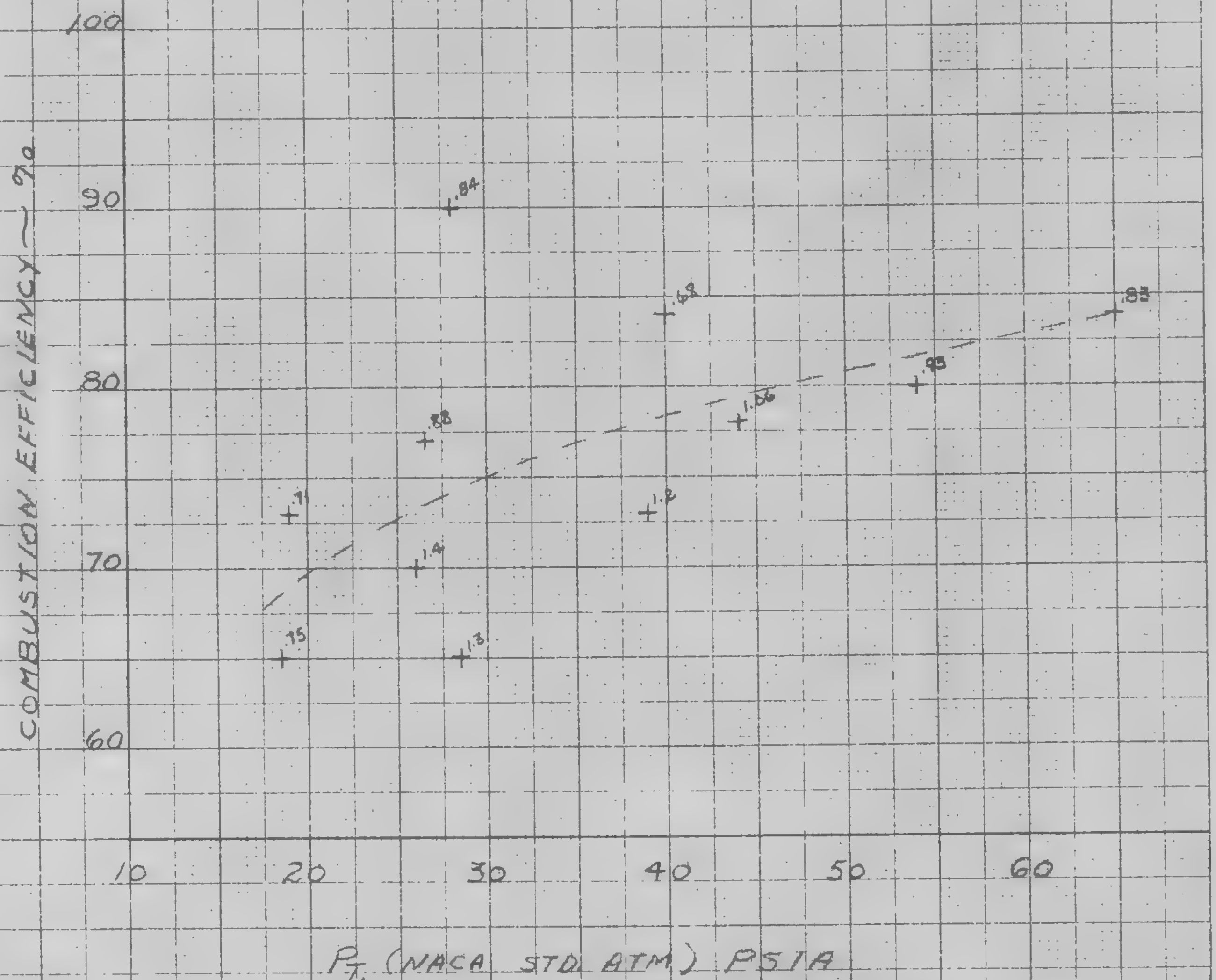
Figure 5. Combustion Efficiency vs. Subsonic Diffuser Inlet Stagnation Pressure for Four (4) Flights of Bumblebee BTV, with Pressures Calculated From Normal Shock Rise and Based on NACA Standard Atmosphere at Flight Altitude. (Numbers Indicate Equivalence Ratio)



B. Flight Test: Excellent correlation between flight performance and laboratory performance of burners has been found; however, flight has been limited to the region where it is possible to adequately duplicate flight conditions in readily available test facilities. The best example of this limitation has been high altitude flight, where ramjets have stopped burning at from 30,000 to 45,000 feet, which is also the general limitation of readily available test equipment.

Flight test is, of course, far too expensive for burner development tests unless the burner is in a pretty advanced stage already.

C. Scale of Burner: Experience in scaling burners has clearly indicated that numerous laboratory tests are required of a new size of combustor before satisfactory operation is approached. For example, when a working 18 inch "Bumblebee" burner was scaled up to 24 inches, about twenty weeks of testing over a period of nine months was required before the 24 inch burner reached the standards of performance of the 18 inch burner. This need for testing requires that considerable investment be made in new facilities, with an attendant long and expensive training, research and development period before ramjets any larger than those currently being developed can be built.

D. Starting Problems: Starting tests in laboratory facilities cannot duplicate all the factors involved, so for this reason a sizeable percentage of loss of valuable flight missiles must be expected in launching failures. Several such failures of 18 inch Bumblebee flights occurred before a proper method of correlating laboratory and flight starting characteristics was found and a solution to the problem devised. The two basic problems of starting are:

1. Ignition under very high combustion velocities, unless a disposable restrictor is used.

2. Delay in developing full thrust, which may mean that the missile decelerates to below a speed from which it can accelerate under its own power.

E. Altitude Limitations: Although this limitation is mentioned under Section III, B, it represents such a complex of unsolved problems that special emphasis is required. The fact that combustor performance deteriorates rapidly at pressures less than one atmosphere presents a problem which is made doubly hard to solve because the basic factors causing this behavior have not been adequately determined. Figure 5 shows the tendency of combustion efficiency to decline with reduced pressure even at pressures greater than one atmosphere. Research and development of this phase has been severely retarded by lack of test facilities leaving the development of a high altitude propulsion unit several years away.

F. Control Problems: Fuel flow in a ramjet is, of course, a function of flight velocity and altitude requiring that sensitive air and fuel measuring and metering devices be developed. The development of such controls has presented a considerable problem in the past, and will become even more involved in the future as ramjets have to operate at a wider range of altitudes.

BOOSTERS FOR LONG RANGE RAMJETS

## BOOSTERS FOR LONG RANGE RAMJETS

Ramjet efficiencies necessary to make long ranges possible can only be achieved around Mach numbers of the order of 3.

Furthermore these efficiencies require the use of diffusers with poor "off design" characteristics.

A self acclerating ramjet would therefore entail the extreme complication of variable inlets variable nozzles and variable combustors and the elaborate development program leading to their design.

Barring this complication the only alternative is to boost the ramjet to a Mach number close to 3. The pressure in the combustion chamber at such a Mach number is about thirty (30) times the ambient pressure. Therefore launching a ramjet at this high a Mach number also means launching it at altitude lest combustion chamber pressures lead to prohibitive structural weight.

To boost a missile of the size required to both high velocity and high altitude the only means available to date is the liquid propelled, turbo pump rocket. All the design problems involved in the construction of such a booster i.e. power plant, stabilization, structural and aerodynamic problems are the same as those that have to be solved for a pure rocket type missile.

In addition to these the ramjet designer is faced with the assignment of solving the same problems - power plant, structure, stabilization and guidance, aerodynamics - at least equally difficult relating to the ramjet proper.

Of these, contrasting with the rocket the power plant problem involves years of development work and millions of dollars of testing equipment.

The construction of the composite missile will involve the building of two different type power plants, two fuel systems, two stabilization and guidance systems. The probability of malfunctioning of these items is also doubled.

In short, due to the necessity of a high altitude high velocity boost the effort required to produce a long range ramjet is roughly twice that for a rocket missile, both from the standpoint of engineering and the standpoint of construction. Comparing ramjet performance to the performance of a rocket with glide path, it would appear that the benefits derived from this extra effort are questionable at best.

# PROPOSED METHOD FOR IMPROVING HIGH ALTITUDE PERFORMANCE OF PRESENT RMI 20,000 POUND THRUST ROCKET

## EFFECT OF NOZZLE EXPANSION ON ROCKET THRUST

In report ZP-48-35003, page 76, mention is made of increasing the missile range by increasing the expansion ratio of the rocket nozzles. It is proposed that this be accomplished without changing the design of the present motor.

The present nozzle, with a 16.63 inch exit diameter, over expands the exhaust up to 5,000 feet and above this altitude expansion is incomplete. By increasing the exit diameter, and allowing more complete expansion at high altitude, rocket thrust can be appreciably increased. However, to prevent severe thrust losses at low altitude, an annuler plug must be inserted in the exit nozzle so that the excess area is filled with a solid resistant material (probably graphite) to prevent over expansion. At 20,000 feet, the plug is released and the thrust from the greater expansion is provided.

Figure 6 shows the thrust increase with altitude for the present nozzle design and the gain that can be expected if greater expansion is used.

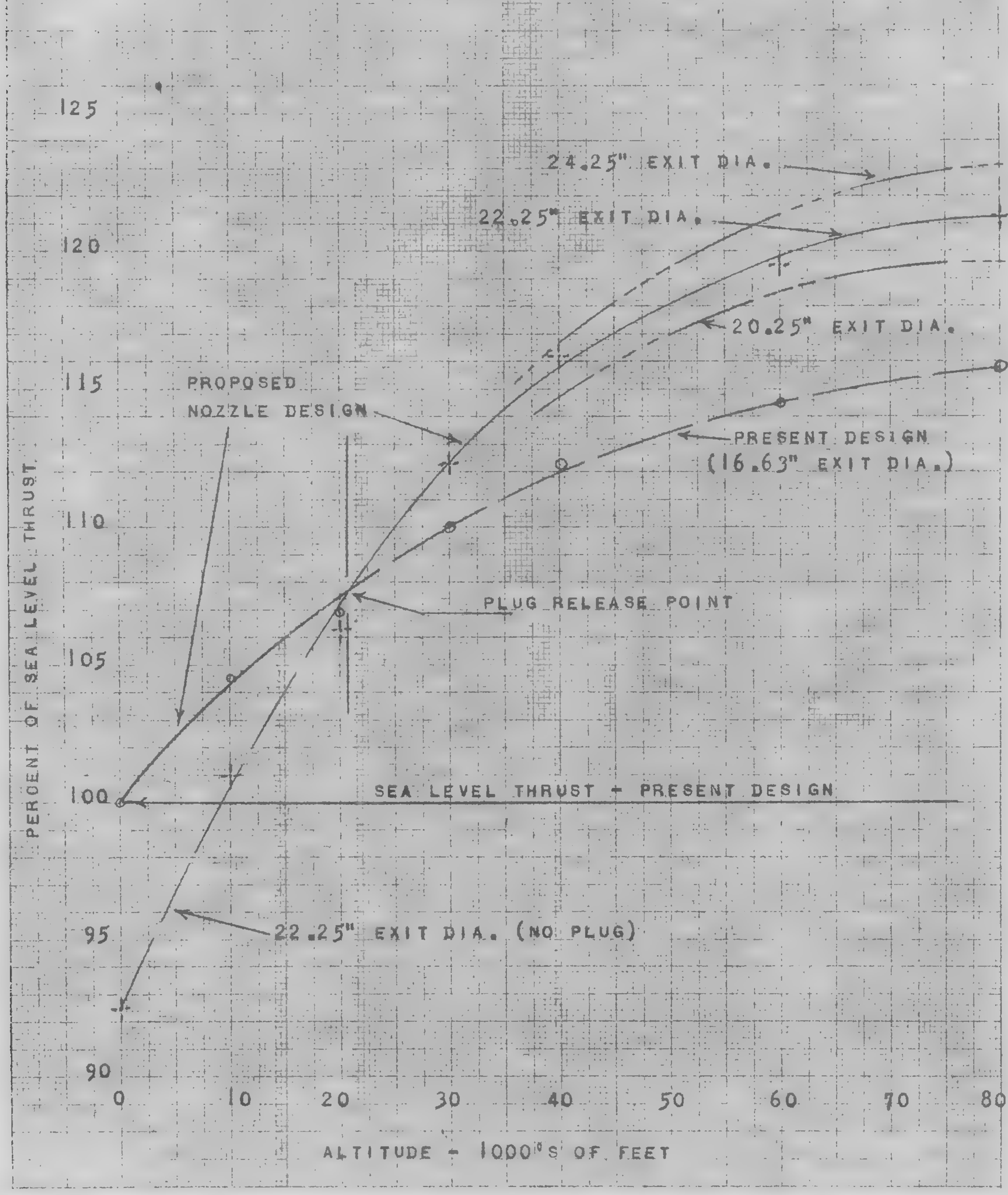
Figure 7 is a schematic drawing of the proposed nozzle extension and plug insert as related to the present rocket design. The only change is an addition to the basic design.

The following table shows the percent gain in thrust by expanding beyond the present exit diameter. The Figures represent the net gain from sea level to 520,000 feet altitude (using a plug during the first 20,000 feet).

### TABLE

| DESIGN | EXIT DIAMETER (INCHES) | PERCENT THRUST GAIN |
|---|---|---|
| #1 (present) | 16.63 | 0 |
| #2 | 20.25 | 3.1 |
| #3 | 22.25 | 4.5 |
| #4 | 24.25 | 6.0 |

FIGURE 6
EFFECT OF NOZZLE DESIGN
ON
ROCKET THRUST



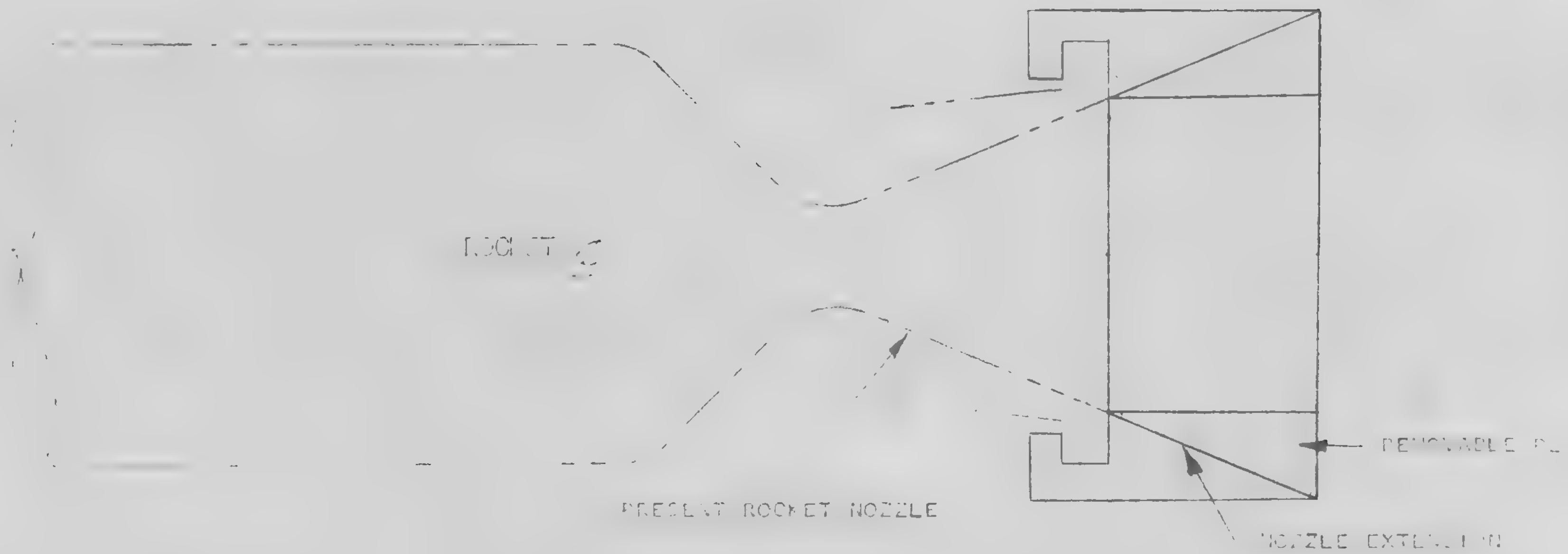



FIG. 7 SCHEMATIC —— PROPOSED ROCKET NOZZLE EXTENSION AND REMOVABLE PLUG

DISCUSSION:

Rocket operation is to be maintained from sea level to 520,000 feet altitude. Information on pressures at high altitude was obtained from NACA report number [illegible] and other references i.e., Rand Project, Whipple, Betz Et Al, and Warfield. Pressure ratios were then determined by using chamber pressure at 300 psia and various altitude pressures. Rocket thrust was calculated by the formula:

$$F = C_F \, P_C \, a_T \quad \text{(ref. *)}$$

where: $F$ = thrust of rocket

$C_F$ = thrust coefficient

$P_C$ = chamber pressure

$a_T$ = nozzle throat area

In each design shown in the above table, the throat area is the same (47.1 sq. in.). In Figure 6 the thrust curves were terminated at 80,000 feet, since above this altitude, the thrust coefficient is constant.

The larger exit diameters give higher thrust values at high altitude at the expense of a low altitude loss. However the use of an annular plug (see Figure 7 ) appears to be a satisfactory solution. The plug is designed to maintain the [illegible] inch diameter of the present nozzle exit up to [illegible] feet. At this altitude it will be released, leaving the expanded nozzle which will give the higher thrust values above this point.

Design number 3 is approximately 6 percent below optimum at 80,000 feet, however a greater nozzle diameter will cause an external (aerodynamic) drag that will probably more than offset the thrust gain. Above 80,000 feet the thrust of a given design will remain constant while optimum thrust increases. Values of optimum thrust above this altitude cannot be found with the available data.

Figure 6 shows thrust values for three possible extensions to the present nozzle. The solid curve labelled "proposed nozzle design" shows the probable best nozzle design for the 1,000 mile missile in its present configuration.

The nozzle extension will not require a redesign of the present rocket and all its heat balance problems. The extension can be made of graphite (or some other heat resistant material) and attached to the present nozzle lip.

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Ref. * California Institute of Technology Reference Text Book "JET PROPULSION" Eq. 5.17.

# WARHEAD SKIN TEMPERATURES

## WARHEAD SKIN TEMPERATURES

Figure 8 shows the temperatures anticipated for the warhead skin for the ballistic and glide path flights. It can be seen from the curves that the maximum temperature in the ballistic warhead occurs shortly before impact. In the glide type warhead the maximum occurs during the first quarter of the glide range. The skin temperature then decreases during the remainder of the flight. The conditions of the glide path are somewhat less severe, lower skin temperatures and a longer heating period alleviating thermal stresses in the skin considerably.

Materials and techniques currently in use are suitable for the design of a warhead skin for a 1000 mile range ballistic path missile, or a 2000 mile range glide path missile.

The method used to determine the data of Figure 8 is essentially that of Convair Report No. DEVF 346-17. This is a step-by-step method including the effects of heating by the boundary layer, dissociation in the boundary layer, radiation by the skin, and the thermal capacity of the skin.

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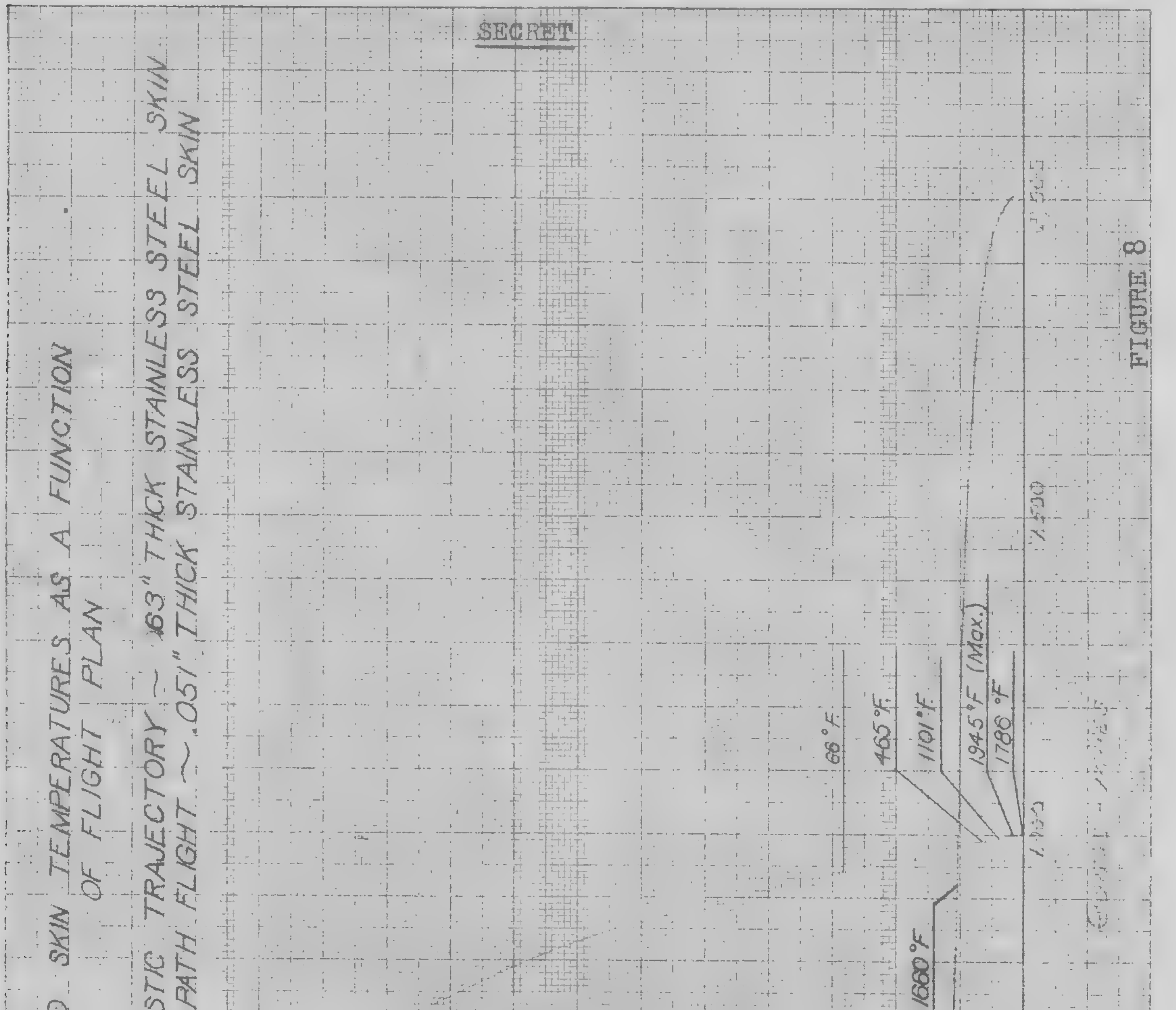


FIGURE 8

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# ROCKET MISSILE OF 2000 MILE RANGE

# WITH

# 6000 POUND GLIDE TYPE WARHEAD

A. General: The 6000 pound warhead 1000 mile ballistic missile described in CVAC Report No. ZP-48-35003 may be considered a first step toward the ultimate development of a glide type warhead missile of roughly double the range.

This possibility for future development was lightly touched upon in Report ZP-48-35003 on pages 2 and 31. Some results of further preliminary study on this problem can now be presented.

Time has not permitted an independent computation of skin friction drags at high Mach numbers during the glide. These computations are very elaborate. Therefore the performance analysis shown is based on the use of skin friction coefficients as determined by RAND.

The establishment of an optimum configuration for the glide type missile was found to be considerably more difficult than for the ballistic type. The warhead size and weight, maximum L/D, altitude and velocity at burnout, warhead skin temperatures are all interdependent parameters. By a process of compromise, a first approximation for an optimum configuration, shown on pages 29 and 30 was arrived at. This configuration can doubtlessly be improved by further study.

B. Description:

| | | |
|---|---|---|
| Overall Length | 83.5 | ft |
| Body Diameter | 6.75 | ft |
| Gross Weight | 74775 | lb |
| Warhead | 6000 | lb |
| Fabricated Wt | 6635 | lb |
| Fuel Wt | 62140 | lb |
| Take off Thrust | 108,000 | lb |
| Range (Approx.) | 2000 | St Miles |

The power plant is the same as that of the proposed ballistic missile including the method of jettisoning the booster stage.

Inasmuch as guidance is necessary during the glide, all intelligence equipment is located in the warhead.

Aluminum construction is used for the alcohol tank section, aft body and empennage. The latter is jettisoned at the end of the booster stage.

The gliding warhead structure separates at burnout and is built out of stainless steel due to temperature considerations. The aft part of this structure is the oxygen tank. The shape of the warhead body is aerodynamically stable at high Mach numbers so that no tail surfaces are required during the glide. Roll stabilization is provided by a pair of rollerons located at the base of the nose cone. During the glide trim in pitch and yaw is controlled by a seiveling (canard type) nose.

C. **Weight Statement:**

GROSS WEIGHT SUMMARY

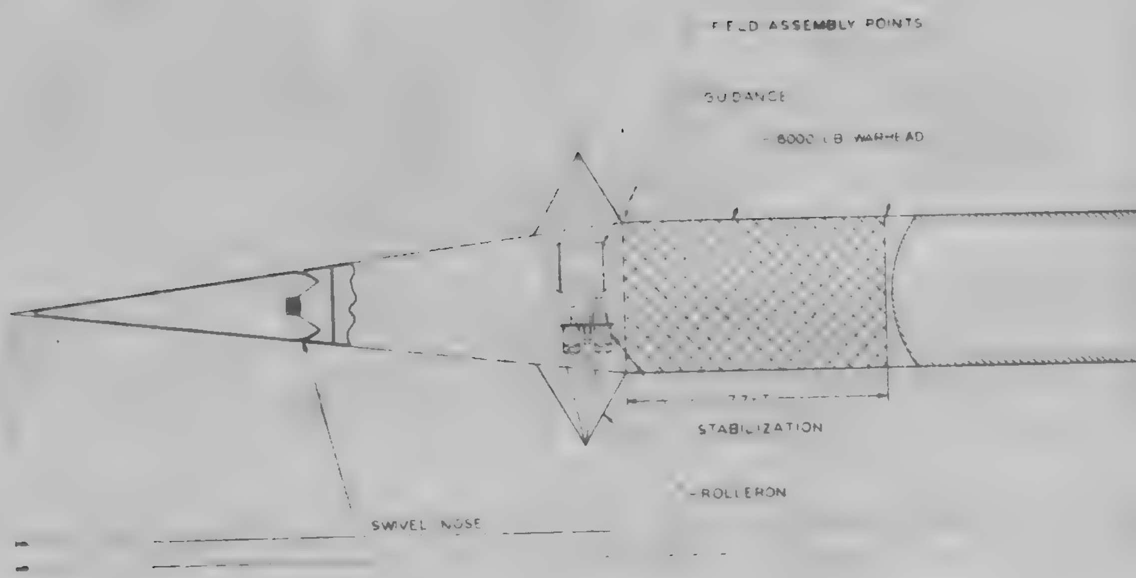
6000 LB GLIDE TYPE WARHEAD

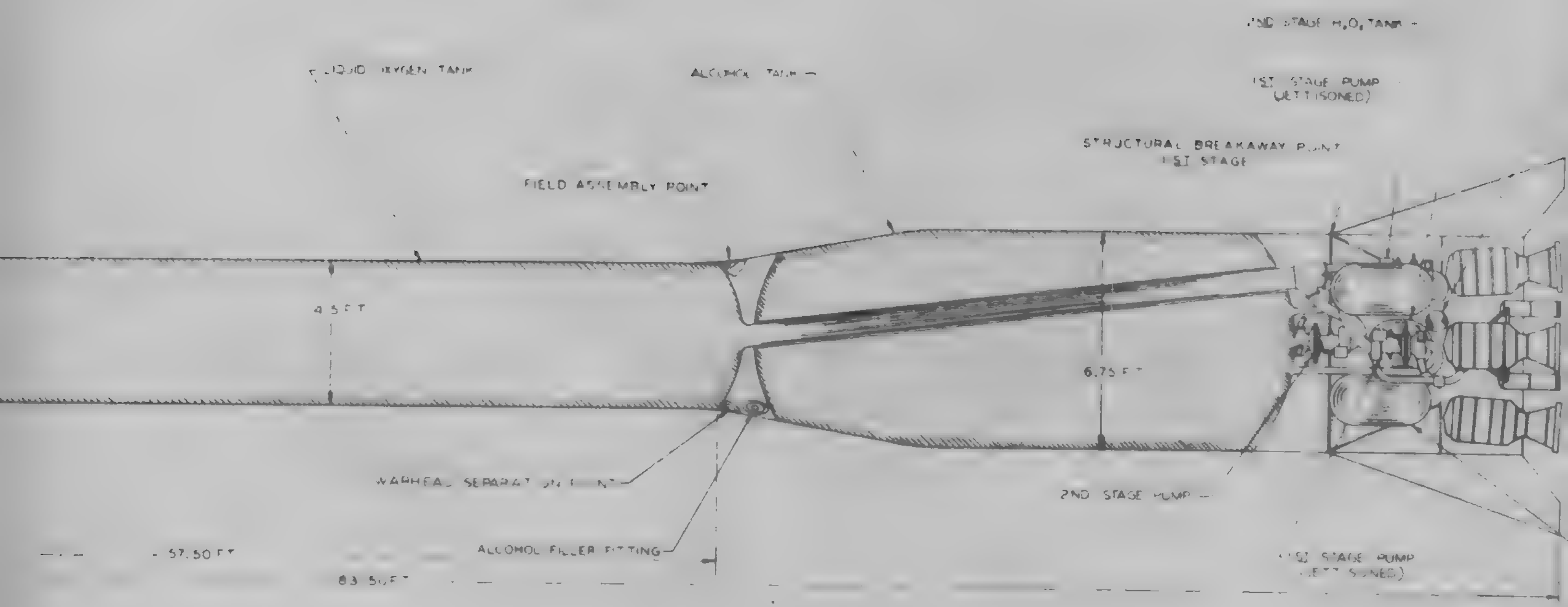
| | STAGE I | | | STAGE II | | |
|---|---|---|---|---|---|---|
| | WEIGHT | ARM | MOMENT | WEIGHT | ARM | MOMENT |
| Gross Weight | 74775 | 626 | 46864560 | 17492 | 555 | 9709568 |
| Weight Empty | ([illegible]) | (515) | (6499560) | (99[illegible]6) | (3[illegible]6) | ([illegible]) |
| Payload | 6000 | 265 | 1590000 | 6000 | 265 | 1590000 |
| Body aft of separation | 799 | 860 | 687400 | 554 | 820 | 453500 |
| Fins | 100 | 989 | 98900 | --- | --- | --- |
| Power plant | 3197 | 946 | 3027360 | 1039 | 945 | 981360 |
| Equipment Nose | 715 | 215 | 154000 | 715 | 215 | 154000 |
| Equipment tail | 349 | 910 | 317500 | 143 | 863 | 123300 |
| Body fwd of separation | 1475 | 423 | 624400 | 1475 | 423 | 624400 |
| Fuel | (62140) | (650) | (40365000) | (7566) | (764) | (5783000) |
| $H_2O_2$ | 1243 | 936 | 1165000 | 151 | 949 | 143000 |
| Lox | 33832 | 507 | 17200000 | 4119 | 670 | 2760000 |
| Alcohol | 27065 | 811 | 22000000 | 3296 | 876 | 2880000 |
| Wt. at end of 1st stage | 60897<br>20201 | 610 | 12282560 | | | |
| Wt. at end of 2nd stage | | | | 9926 | 396 | 3926560 |
| Warhead weight | | | | 8190 | 290 | 2368400 |

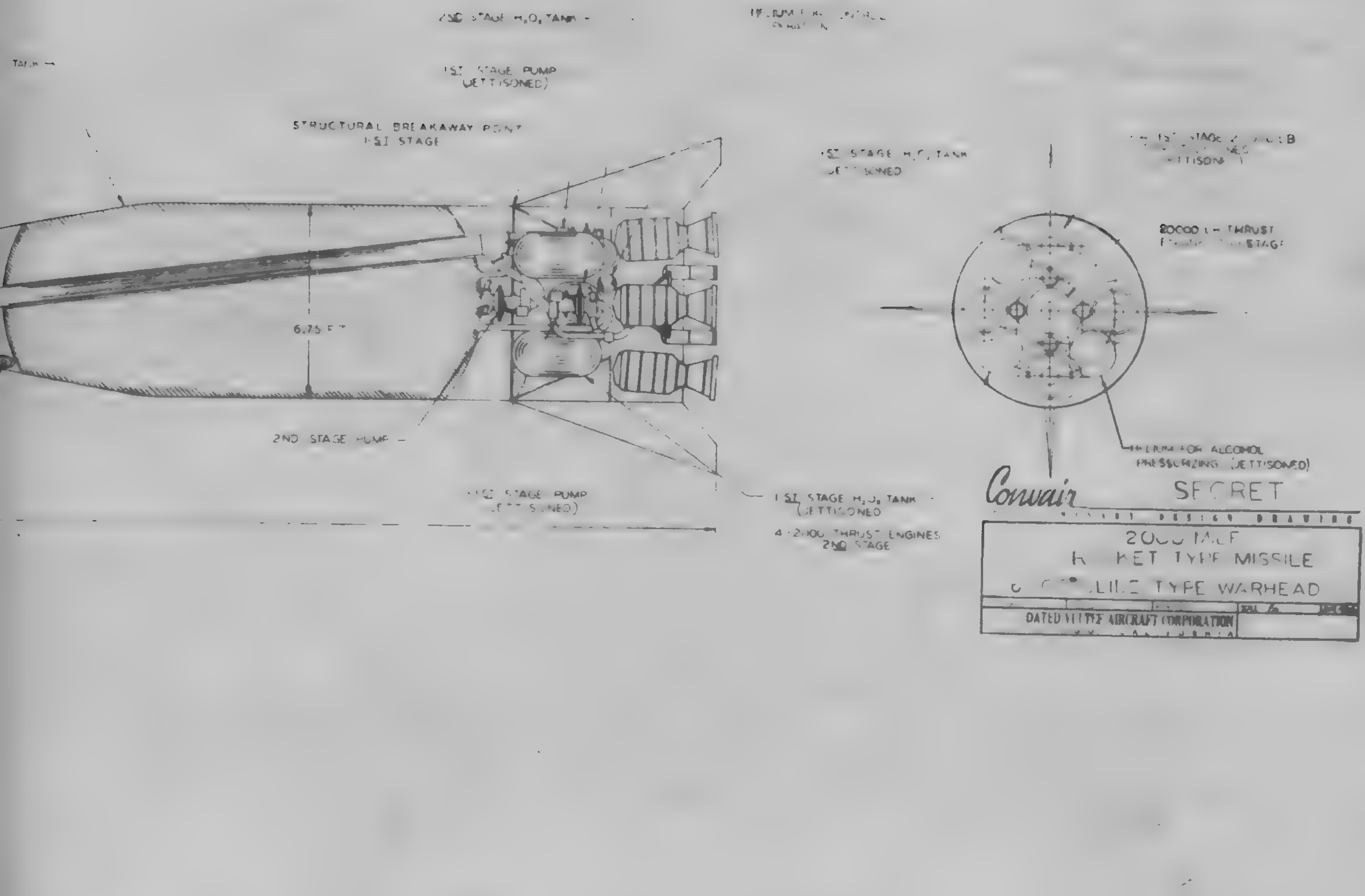
WEIGHT EMPTY
WEIGHT AND BALANCE
6000 lb Glide Type Warhead

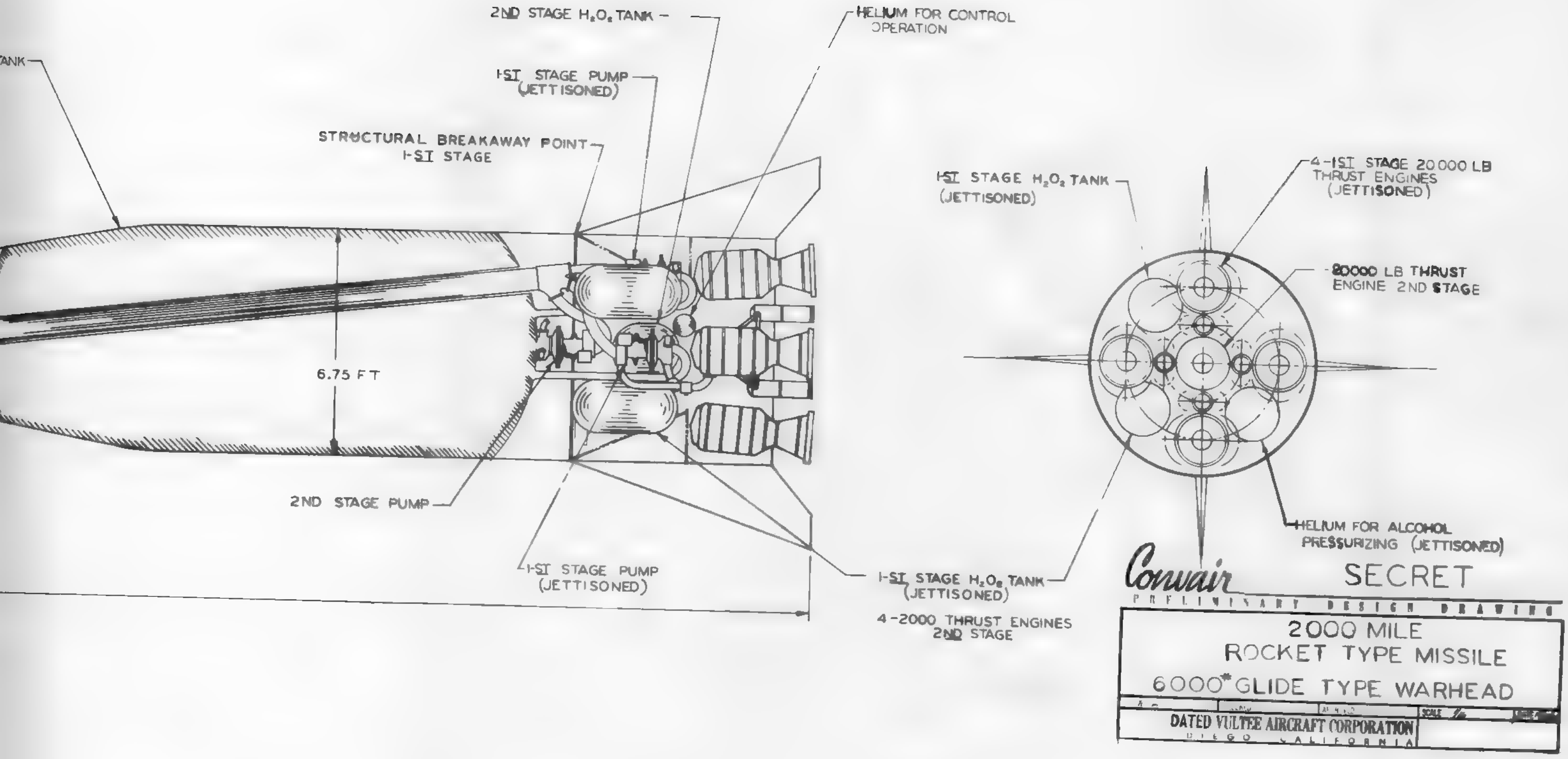
| ITEM | STAGE I WEIGHT | STAGE I ARM | STAGE I MOMENT | STAGE II WEIGHT | STAGE II ARM | STAGE II MOMENT |
|---|---|---|---|---|---|---|
| Glide Warhead | (8190) | (290) | (2368400) | (8190) | (290) | (2368400) |
| Nose Cone | 269 | 146 | 95000 | 269 | 146 | 95000 |
| Warhead | 6000 | 265 | 1590000 | 6000 | 265 | 1590000 |
| Fixed EQ + Guid | 545 | 182 | 99000 | 545 | 182 | 99000 |
| Trimming Dev. | 100 | 109 | 10900 | 100 | 109 | 10900 |
| Skin | 1138 | 435 | 495000 | 1138 | 435 | 495000 |
| Residual Fuel | 50 | 680 | 34000 | 50 | 680 | 34000 |
| Fittings | 20 | 507 | 10100 | 20 | 507 | 10100 |
| Tank Ends | 68 | 507 | 34400 | 68 | 507 | 34400 |
| Power Plant | (3197) | (946) | (3027360) | (1039) | (945) | ( 981360) |
| Motor & Prop Valves | (1225) | (974) | (1192600) | ( 457) | (975) | ( 445600) |
| (4) 20000# Thrust | 768 | 973 | 747000 | --- | --- | --- |
| (1) 20000# Thrust | 192 | 973 | 187000 | 192 | 973 | 187000 |
| (4) 2000 # Thrust | 165 | 988 | 163000 | 165 | 988 | 163000 |
| (4) 20000# Thrust Prop Valve | 80 | 956 | 76500 | 80 | 956 | 76500 |
| (1) 20000# Thrust Prop Valve | 20 | 956 | 19100 | 20 | 956 | 19100 |
| Pumps | (725) | (935) | ( 679000) | ( 145) | (916) | ( 133000) |
| (4) 20000 Thrust | 580 | 941 | 546000 | --- | --- | --- |
| (1) 28000 Thrust | 145 | 916 | 133000 | 145 | 916 | 133000 |
| Mounts | ( 210) | (935) | ( 196600) | ( 110) | (931) | ( 102500) |
| 80000 Mount | 100 | 941 | 94100 | --- | --- | --- |
| 28000 Mount | 110 | 931 | 102500 | 110 | 931 | 102500 |
| Valves and Regulators | ( 160) | (941) | ( 150600) | ( 40) | (941) | ( 37600) |
| 28000 | 40 | 941 | 37600 | 40 | 941 | 37600 |
| 80000 | 120 | 941 | 113000 | --- | --- | --- |

| | STAGE I | | | STAGE II | | |
|---|---|---|---|---|---|---|
| | WEIGHT | ARM | MOMENT | WEIGHT | ARM | MOMENT |
| Lines and Fittings | ( 365) | (926) | ( 338000) | ( 135) | (919) | ( 124200) |
| 28000 | 95 | 925 | 88000 | 95✓ | 925 | 88000 |
| 80000 | 270✓ | 925 | 250000 | 40 | 905 | 36200 |
| Press Supply | ( 150)✓ | (932) | ( 139860) | ( 35)✓ | (924) | ( 32360) |
| Heat Exchanger - Oxygen | 12 | 925 | 11100 | 12 | 925 | 11100 |
| Helium Tank Alcohol Pressure | 115 | 936 | 107500 | --- | --- | --- |
| Helium Tank Control Operation | 13 | 916 | 11900 | 13 | 916 | 11900 |
| Helium | 10 | 936 | 9360 | 10 | 936 | 9360 |
| Pump Prop. System | ( 132) | (937) | ( 123700) | ( 32)✓ | (941) | ( 30100) |
| (2) $H_2O_2$ Tank - Large | 100✓ | 936 | 93600 | --- | --- | --- |
| (1) $H_2O_2$ Tank - Small | 32 | 941 | 30100 | 32 | 941 | 30100 |
| Residual Fuel | ( 230) | (902) | ( 207000) | ( 85) | (895) | ( 76000) |
| 28000 | 60 | 902 | 54000 | 60 | 902 | 54000 |
| 80000 | 170 | 902 | 153000 | 25 | 880 | 22000 |
| Fixed Equipment | ( 349) | (910) | ( 317500) | ( 143) | (863) | ( 123300) |
| (6) Valves | 36 | 906 | 32600 | 18 | 906 | 16300 |
| (6) Response Units | 30 | 906 | 27200 | 15 | 906 | 13600 |
| 8000 Swivel Cont. | 40 | 976 | 39000 | 40 | 976 | 39000 |
| **80000 Swivel Cont.** | 153 | 946 | **145000** | --- | --- | --- |
| Staging Controls | 40 | 966 | 38600 | 20 | 966 | 19300 |
| Warhead Eject. | 50 | 702 | 35100 | 50 | 702 | 35100 |
| Aft Body | ( 7[illegible]) | (860) | ( 697400) | ( 554) | (818) | ( 45[illegible]) |
| **Alc Tank** | 392 | 811 | **318000** | 392 | 811 | 318000 |
| Aft Bulkhead | 39 | 902 | 35200 | 39 | 902 | 35200 |
| Fwd Bulkhead | 23 | 710 | 16200 | 23 | 710 | 16200 |
| Lox. Tunnel | 60 | 800 | 48000 | 60 | 800 | 48000 |
| Engine Struct. | 285 | 946 | 270000 | 40 | 906 | 36100 |
| Fin | [illegible] | [illegible] | [illegible] | --- | --- | --- |

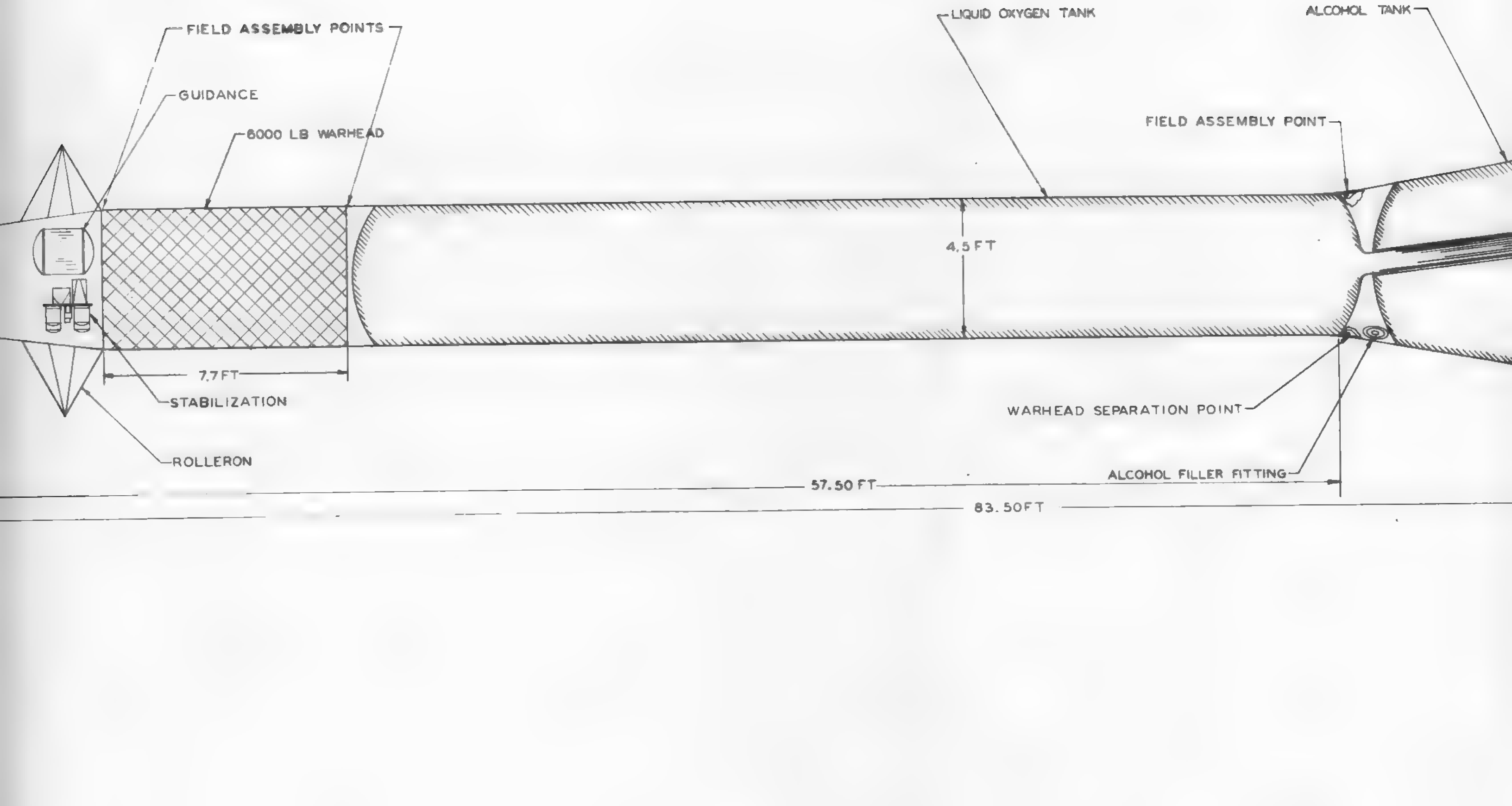

FIELD ASSEMBLY POINTS
GUIDANCE
8000 LB WARHEAD
STABILIZATION
ROLLERON
SWIVEL NOSE


LIQUID OXYGEN TANK
ALCOHOL TANK
1ST STAGE PUMP
STRUCTURAL BREAKAWAY POINT
1ST STAGE
FIELD ASSEMBLY POINT
4.5 FT
6.75 FT
2ND STAGE PUMP
ALCOHOL FILLER FITTING
57.50 FT
1ST STAGE PUMP

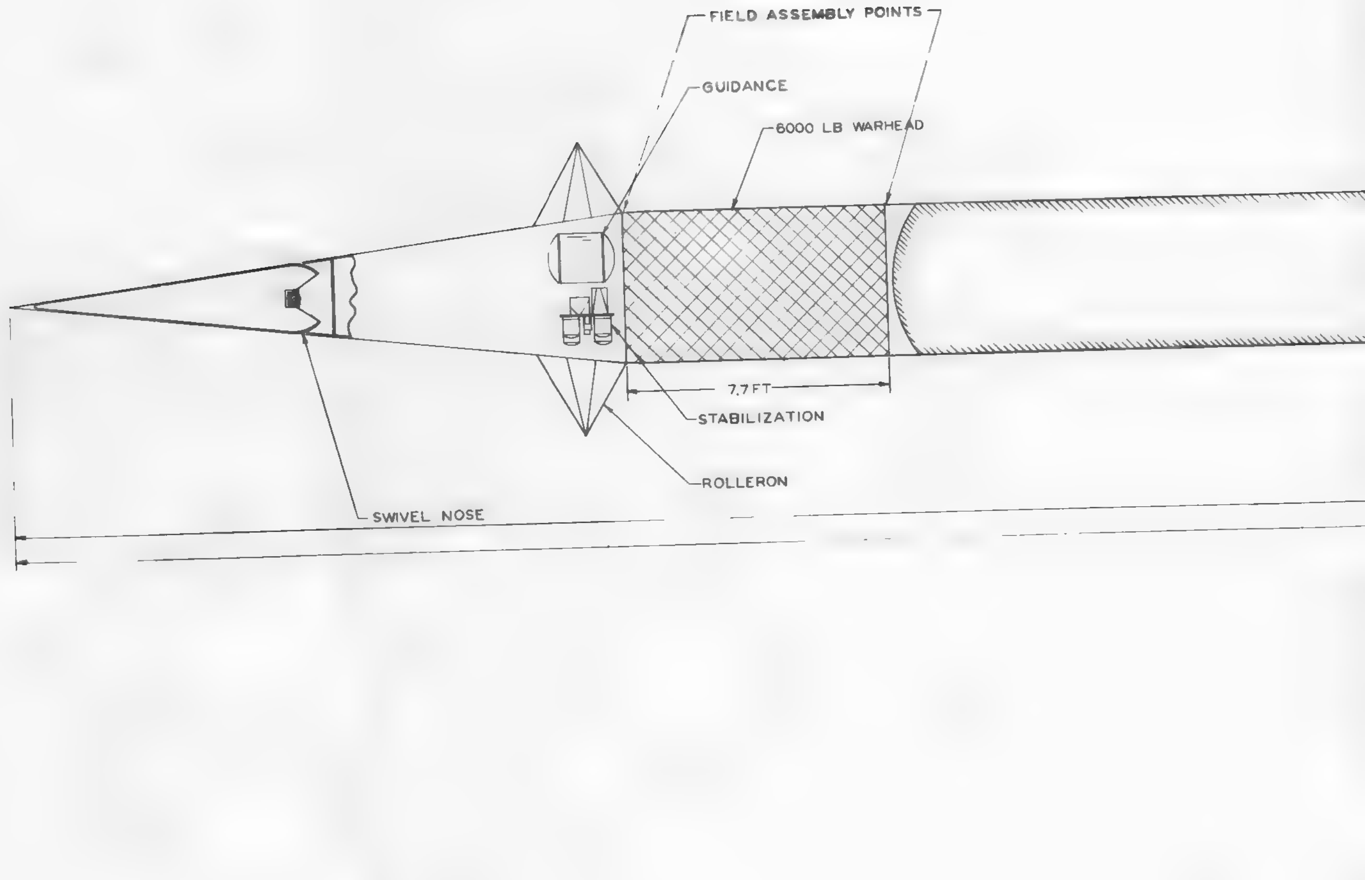

STRUCTURAL BREAKAWAY POINT
1ST STAGE
1ST STAGE PUMP
(JETTISONED)
6.75 FT
2ND STAGE PUMP
1ST STAGE H2O2 TANK
(JETTISONED)
4-2000 THRUST ENGINES
2ND STAGE
20000 LB THRUST
HELIUM FOR ALCOHOL
PRESSURIZING (JETTISONED)
Convair
SECRET
TYPE WARHEAD


ANK
2ND STAGE $H_2O_2$ TANK
HELIUM FOR CONTROL OPERATION
1-ST STAGE PUMP (JETTISONED)
STRUCTURAL BREAKAWAY POINT 1-ST STAGE
6.75 FT
2ND STAGE PUMP
1-ST STAGE PUMP (JETTISONED)
1-ST STAGE $H_2O_2$ TANK (JETTISONED)
4-2000 THRUST ENGINES 2ND STAGE
1-ST STAGE $H_2O_2$ TANK (JETTISONED)
4-1ST STAGE 20,000 LB THRUST ENGINES (JETTISONED)
20000 LB THRUST ENGINE 2ND STAGE
HELIUM FOR ALCOHOL PRESSURIZING (JETTISONED)
Convair
SECRET
PRELIMINARY DESIGN DRAWING
2000 MILE
ROCKET TYPE MISSILE
6000# GLIDE TYPE WARHEAD
DATED VULTEE AIRCRAFT CORPORATION
DIEGO CALIFORNIA

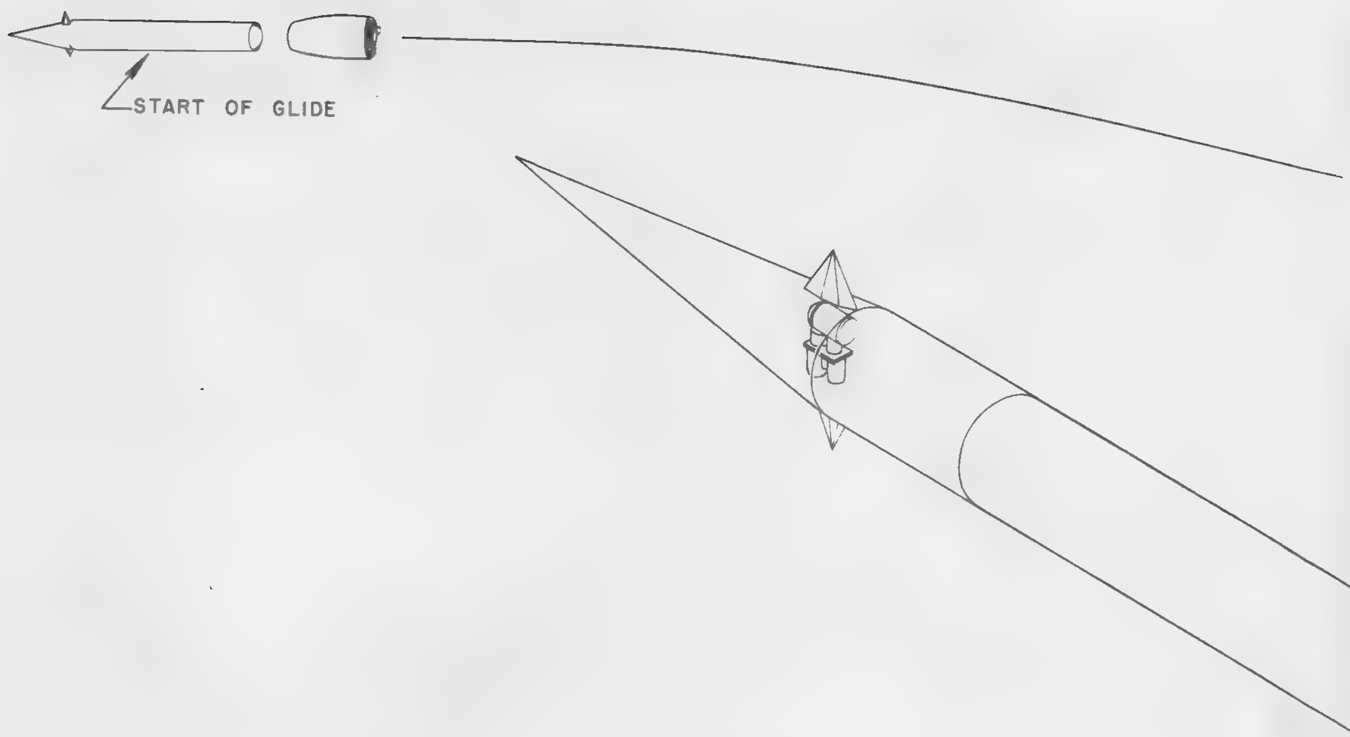
SECRET

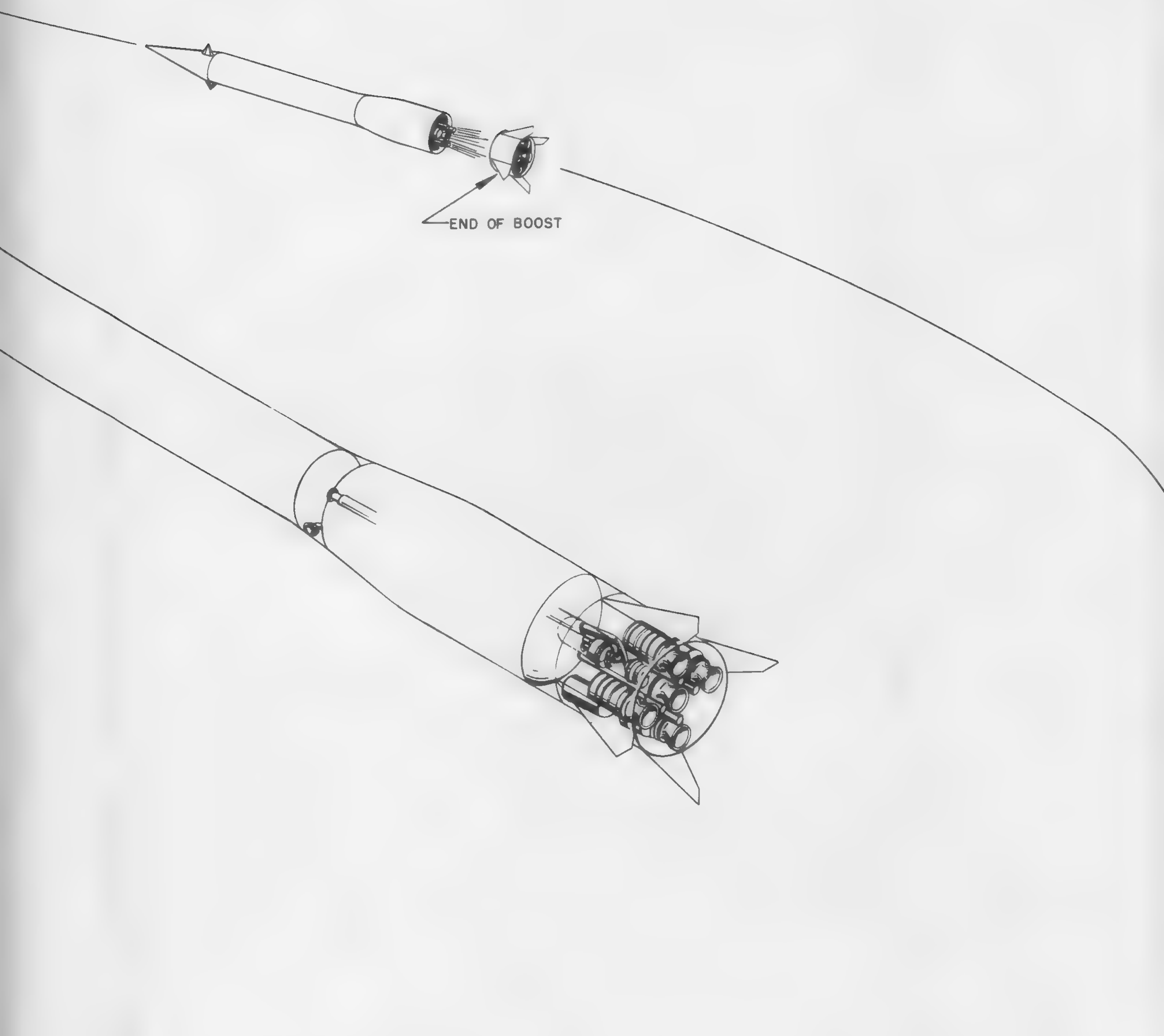



SECRET



FIELD ASSEMBLY POINTS
GUIDANCE
6000 LB WARHEAD
7.7 FT
STABILIZATION
ROLLERON
SWIVEL NOSE


START OF GLIDE


END OF BOOST


SECRET

D. Stability and Control: The glide type rocket missile is stabilized by four (4) stabilizing surfaces (fins) during the booster stage portion of the flight. Only very small fins (12 square feet per panel) are required to stabilize the missile during the booster stage and no fins are required during the final stage of burning. This is a result of the excellent stabilizing characteristics of the body along and the forward position of the center of gravity due to the location of the warhead near the nose.

During powered flight the missile is controlled by the use of swiveling rockets in the same manner as noted on page 12 of reference 1.

The glide configuration will be controlled in pitch and yaw by a swiveling (canard type) nose. This will maintain the necessary angle of attack for maximum L/D and keep the missile on the correct azimuth. This appears to be the best type of control in pitch and yaw, however, additional research will be required on this type of control to ascertain the effectiveness of the nose swiveling and its effect on aerodynamic characteristics of the rest of the body at supersonic speeds. It is possible that movable surfaces (fins) may be required to maintain the glide configuration at an angle of attack.

In order to control the glide configuration in roll, a set of roll flippers were placed just forward of the warhead (see page 29). Since no large disturbances in roll are anticipated the resultant roll flippers are correspondingly small.

Lift and Center of Pressure of Powered Missile Configuration:

Body Alone: The lift and moment characteristics for the body alone were determined by the method outlined in reference 2 for the low Mach numbers ($M = 1.25$ to $M = 5.0$) and by use of Newtonian flow characte istics reference 3) for $M > 10.0$. The results obtained using the method shown in reference 2 have shown good agreement with German wind tunnel pressure distribution test on the V-2 and static stability wind tunnel tests on the MX-774 at Aberdeen, at low angles of attack ($\alpha < 10^{\circ}$). This theory however, breaks down in the vicinity of $M = 5.0$ for normal nose angles. The next logical step then appears to be Newtonian flow characteristics (reference 3) which appear to check the exact gas dynamic theory (reference 3) in the region of $M = 10.0$ to $20.0$.

Since it is believed that the above two theories hold appreciably well for their respective Mach numbers regions and since it is further believed that no discontinuity will occur in the intermediate region (i.e., $M = 5.0$ to $M = 10.0$), the two curves were faired into each other. A plot of normal force and center of pressure characteristics are presented on page 33 for the complete powered configuration.

Fin Alone: The lift and moment characteristics of the fins were obtained from the method outlined in reference 2 for all Mach number regions in which fins were used (M 7). These results are presented in page .

Complete Configuration: Given the body alone and fin alone lift and moment characteristics above the center of pressure of the complete powered configurations are calculated by

$$\frac{X}{\lambda} = \frac{\frac{dCN_B}{d\alpha}\left(1-\frac{XB}{\lambda}\right) + 2\,\frac{dCN_F}{d\alpha}\,\frac{XF}{\lambda}}{\frac{dCN_B}{d\alpha} + 2\,\frac{dCN_F}{d\alpha}}$$

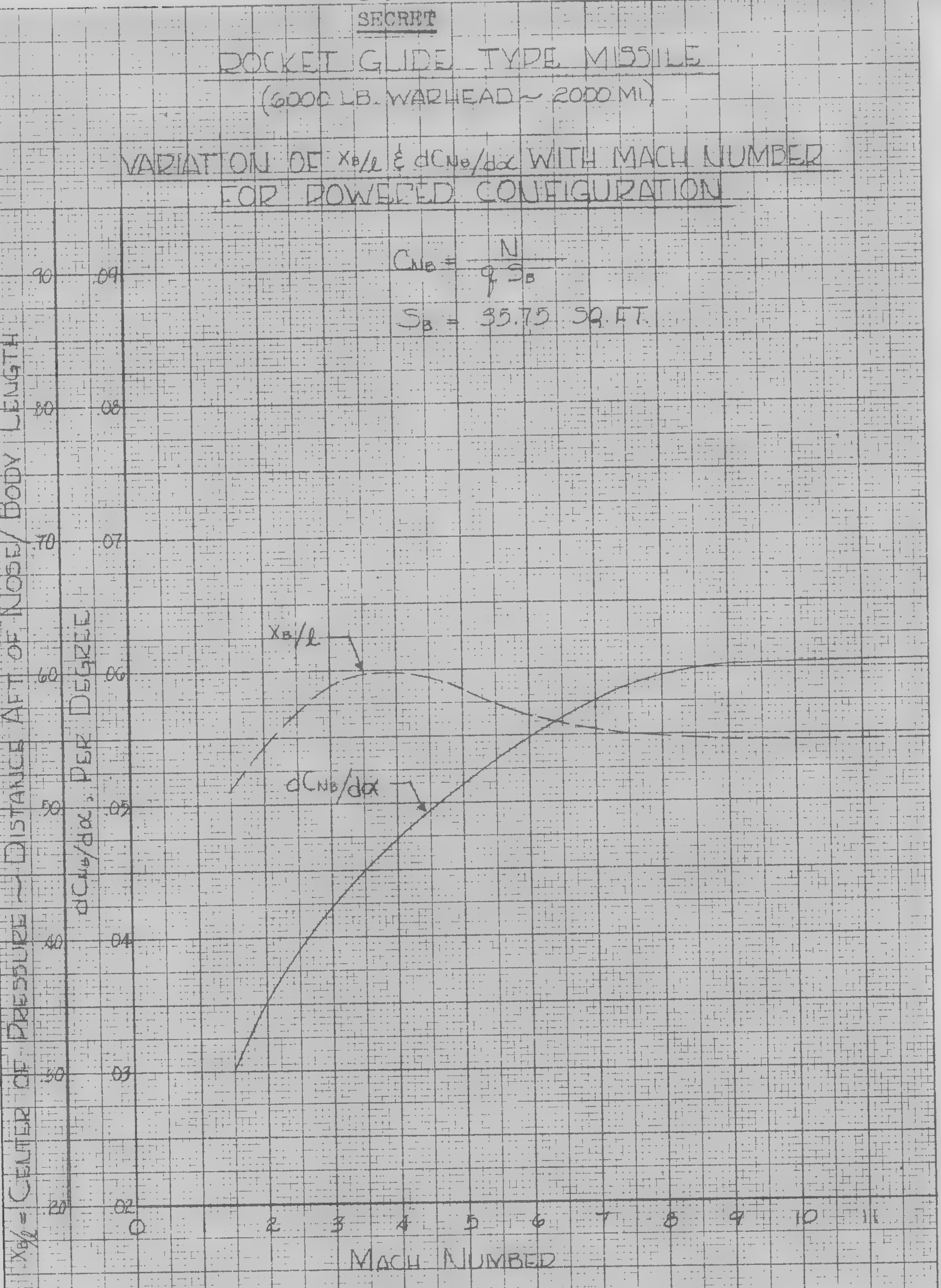
The above equation is used only for the booster stage (Stage I) where fins are necessary. The final stage (Stage II) is stable without fins and thus only body alone characteristics were used.

A plot of the center of pressure and center of gravity versus Mach number of the complete powered configuration for Stages I and II is presented on page. .

Lift and Center of Pressure of Glide Configuration: The glide configuration has a conical nose (half nose angle = 5.0°) and a cylindrical afterbody (see page 29) of 8.7 calibers.

The lift and moment characteristics were obtained by the same method previously outlined for the body alone of the powered flight configuration. However, the results were obtained from higher angle of attack considerations for the glide configuration due primarily to the angles of attack required to maintain maximum Lift/Drag in the glide path.

A plot of $\frac{dCN_G}{d\alpha}$ and $\frac{X_G}{\lambda}$ versus Mach number for the glide configuration is presented on page 35.

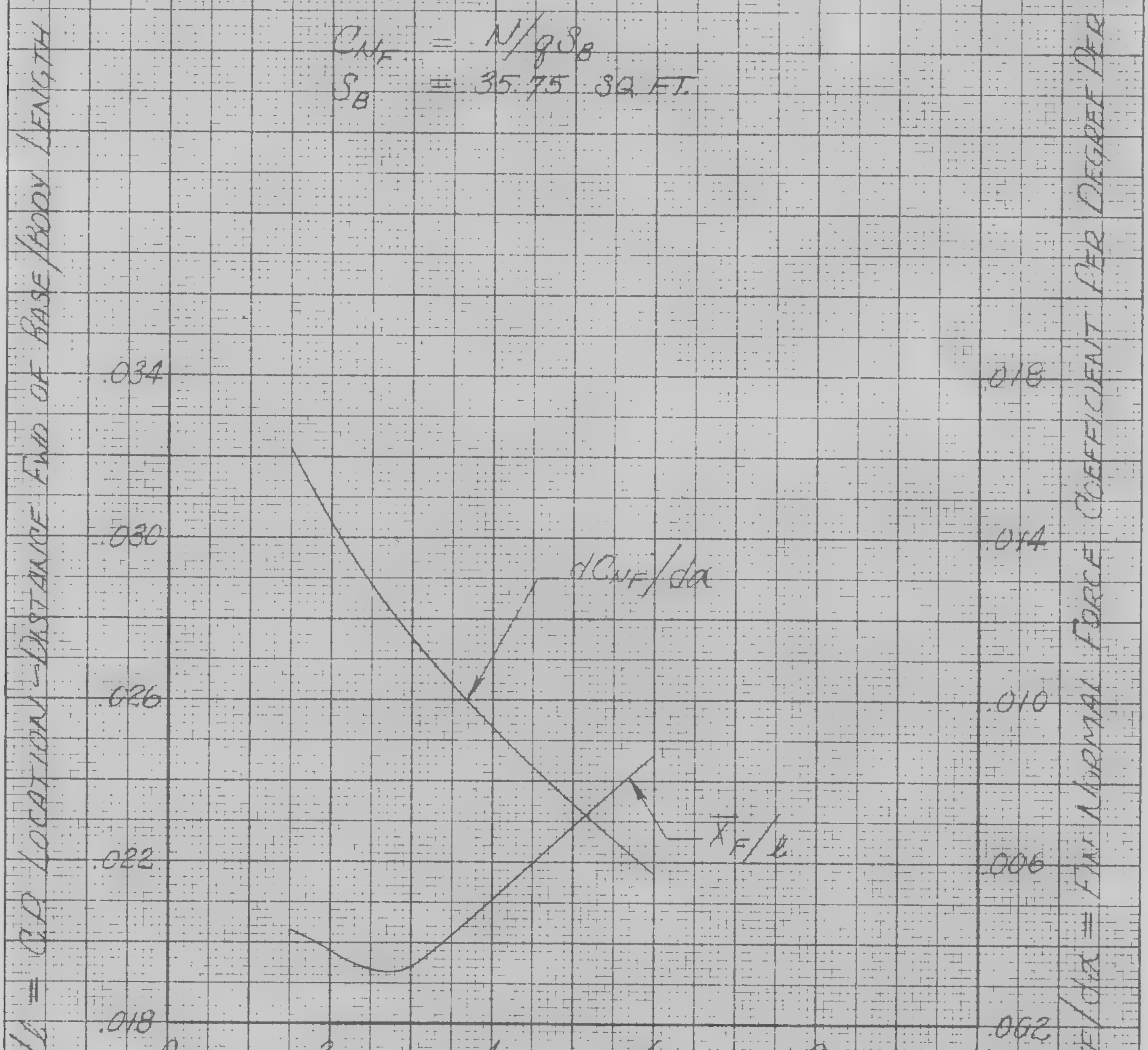

SECRET
ROCKET GLIDE TYPE MISSILE
(6000 LB. WARHEAD ~ 2000 MI.)
VARIATION OF $X_B/\ell$ & $dC_{N_B}/d\alpha$ WITH MACH NUMBER
FOR POWERED CONFIGURATION
$C_{N_B} = \frac{N}{q S_B}$
$S_B = 35.75$ SQ.FT.
$X_B/\ell$
$dC_{N_B}/d\alpha$
$X_B/\ell$ = CENTER OF PRESSURE ~ DISTANCE AFT OF NOSE / BODY LENGTH
$dC_{N_B}/d\alpha$, PER DEGREE
.90 .09
.80 .08
.70 .07
.60 .06
.50 .05
.40 .04
.30 .03
.20 .02
0 2 3 4 5 6 7 8 9 10 11
MACH NUMBER

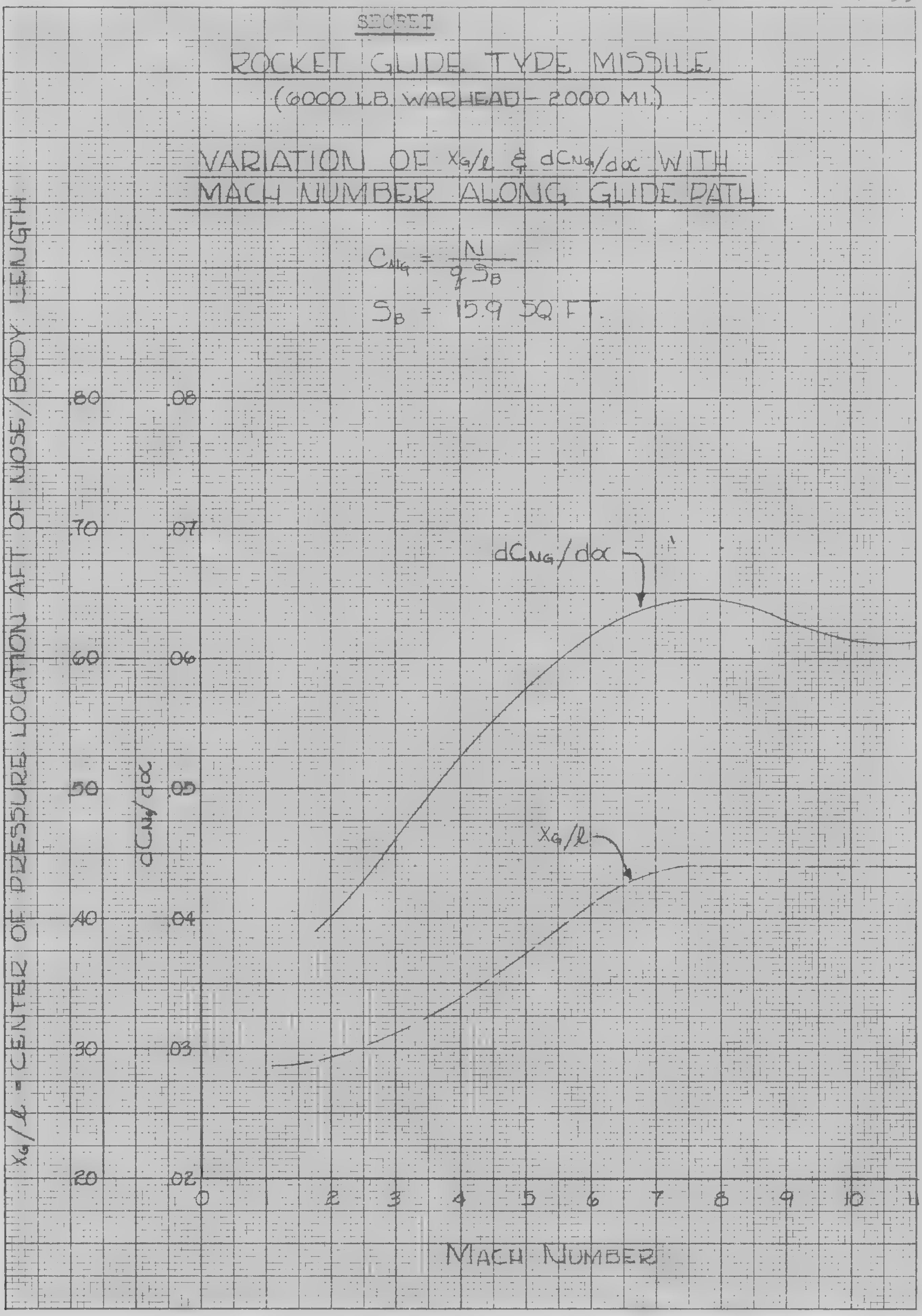
SECRET

ROCKET GLIDE TYPE MISSILE

(6000 LB. WARHEAD – 2000 MI.)

VARIATION OF $dC_{NF}/d\alpha$ & $\bar{X}_F/\ell$ WITH MACH NUMBER FOR FIN ALONE



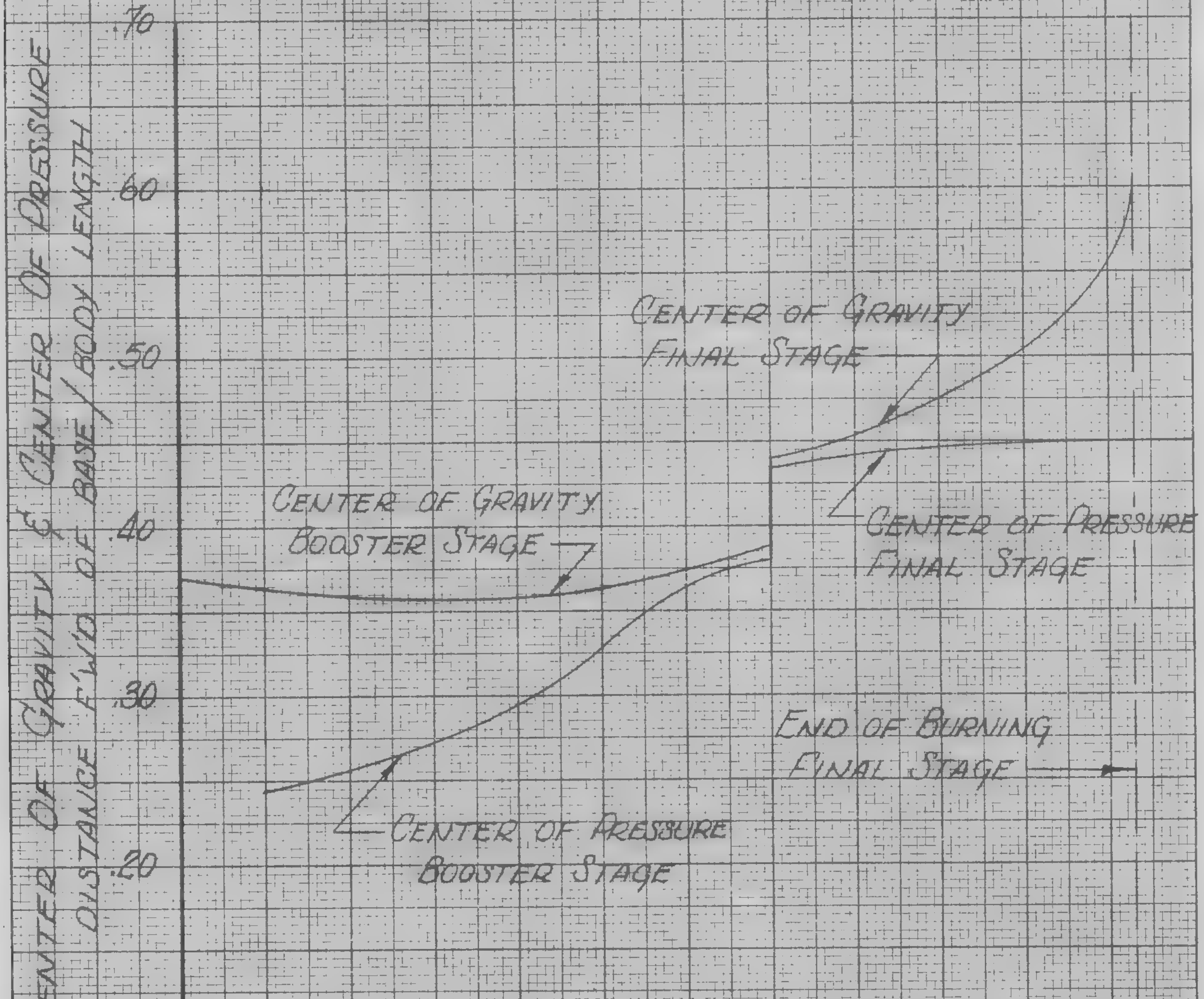

SECRET
ROCKET GLIDE TYPE MISSILE
(6000 LB. WARHEAD - 2000 MI.)
VARIATION OF $X_G/\ell$ & $dC_{NG}/d\alpha$ WITH MACH NUMBER ALONG GLIDE PATH
$C_{NG} = \frac{N}{q S_B}$
$S_B$ = 15.9 SQ FT.
$X_G/\ell$ - CENTER OF PRESSURE LOCATION AFT OF NOSE / BODY LENGTH
$dC_{NG}/d\alpha$
$dC_{NG}/d\alpha$
$X_G/\ell$
.80
.70
.60
.50
.40
.30
.20
.08
.07
.06
.05
.04
.03
.02
0
1
2
3
4
5
6
7
8
9
10
11
MACH NUMBER

SECRET

ROCKET GLIDE TYPE MISSILE

(6000 LB. WARHEAD – 2000 MI.)

VARIATION OF CENTER OF PRESSURE AND CENTER OF GRAVITY LOCATION WITH MACH NUMBER FOR POWERED FLIGHT



E. Performance: The 6000 pound warhead, 2000 mile glide type rocket missile is powered by five (5) 20,000 pound thrust rocket motors and four (4) 2000 pound thrust rocket motors. The missile operates in two stages and is unique in the fact that the same fuel tanks are used in both stages, thus saving much airframe weight. At the end of the booster stage, the four (4) large outside rocket motors are dropped along with related accessories and the entire fin and an appreciable amount of skin and structure. The missile then continues on to the end of burning powered by one 20,000 pound thrust rocket motor and four (4) 2000 pound thrust rocket motors.

At the end of burning of the final stage, the alcohol tank and remaining rocket motors are dropped and thus the remainder of the missile becomes the glide configuration (see page 29 ).

Powered Flight: A series of step-by-step calculations were conducted in order to obtain the correct altitude and velocity along a horizontal trajectory at the end of burning such that the glide configuration would be in equilibrium at the maximum L/D at this point.

The missile was launched vertically for 20 seconds and then assumed an angle of $77^\circ$ with respect to the horizontal. The remainder of the flight was along a zero angle of attack path.

The step-by-step calculations were based on the following relationships which are based on the assumption of a zero angle of attack flight path.

$$\frac{dV}{dt} = g\left[\sin\gamma + \frac{T - D}{W}\right]$$

$$\frac{d\gamma}{dt} = -\frac{g\cos\gamma}{V}$$

$$\frac{dh}{dt} = V\sin\gamma$$

$$\frac{d\,R_G}{dt} = V\cos\gamma$$

Where:

T = thrust for a given altitude

$D = \text{Drag} = q\,S_B\,C_D$

W = Weight at a given time

$\gamma$ = Flight path angle with respect to the horizontal plane at launching site

V = Altitude at a given time

$R_G$ = Range at a given time

t = Time

The variation of thrust with altitude was calculated by increasing the basic sea level thrust of 108,000 pounds for Stage I and 28,000 pounds for Stage II respectively by the percentage indicated in page 18.

The drag analysis used for calculating wave drag and base drag was based on the method outlined in reference 2 for all Mach numbers. These values are identical to those presented in reference 3. The skin friction drag was determined by reference 2 up to M = 2.0 and faired into the values obtained from reference 3 at M = 7.0. Sufficient time did not permit a complete analysis of the method of reference 3 for calculating skin friction and thus the values used at high Mach numbers were taken directly from the RAND report (reference 3) and adjusted to the present glide missile. A plot of the total drag coefficient versus Mach number for the complete powered configuration is presented on page 41 and for the glide configuration on page 42.

<u>Glide Path Flight:</u> The range of the glide path portion of the flight was calculated by the following relationships (see reference 4).

$$g = \frac{V^2}{R} + g\frac{L}{W} \qquad (1)$$

$$R_{glide} = \frac{1}{2g}\int_{U_{E_1}}^{U_{E_f}} \frac{L}{D}\frac{1}{1-\frac{V^2}{gR}}\,dU_E \qquad (2)$$

Where: $U_{E_1} = V_1^2 + 2gh_1$ = Initial Energy

$U_{E_f} = V_f^2 + 2gh_f$ = Final Energy

L/D = lift/drag

W = Weight at a given time

$\gamma$ = Flight path angle with respect to the horizontal plane at launching site

V = Altitude at a given time

$R_G$ = Range at a given time

t = Time

The variation of thrust with altitude was calculated by increasing the basic sea level thrust of 108,000 pounds for Stage I and 28,000 pounds for Stage II respectively by the percentage indicated in page 18.

The drag analysis used for calculating wave drag and base drag was based on the method outlined in reference 2 for all Mach numbers. These values are identical to those presented in reference 3. The skin friction drag was determined by reference 2 up to M = 2.0 and faired into the values obtained from reference 3 at M = 7.0. Sufficient time did not permit a complete analysis of the method of reference 3 for calculating skin friction and thus the values used at high Mach numbers were taken directly from the RAND report (reference 3) and adjusted to the present glide missile. A plot of the total drag coefficient versus Mach number for the complete powered configuration is presented on page 41 and for the glide configuration on page 42.

Glide Path Flight: The range of the glide path portion of the flight was calculated by the following relationships (see reference 4).

$$g = \frac{V^2}{R} + g\,\frac{L}{W} \qquad (1)$$

$$R_{glide} = \frac{1}{2g}\int_{U_{E_i}}^{U_{E_f}} \frac{L}{D}\,\frac{1}{1 - \frac{V^2}{gR}}\,dU_E \qquad (2)$$

Where: $U_{E_i} = V_i^2 + 2gh_i$ = Initial Energy

$U_{E_f} = V_f^2 + 2gh_f$ = Final Energy

L/D = lift/drag

$R$ = radius of earth

$g$ = gravitational acceleration

$W$ = weight

$V$ = velocity at any time

$V_i$ = velocity at start of glide

$V_t$ = velocity at end of glide

$R_{glide}$ = range of glide path

From equation (1) the required altitude may be determined to attain equilibrium at the angle of attack for maximum L/D using a given weight and velocity at the end of burning.

Equation (2), in turn, will give the range traversed during the glide path portion of the flight.

A plot of the glide path portion of the flight calculated by this method is presented on page 23.

The values of L/D used in the glide path calculations were obtained by

$$L/D = \frac{\frac{dC_L}{d\alpha}\alpha}{C_{Do} + \frac{dC_N}{d\alpha}\alpha^2}$$

Where:

$C_L = C_N \cos \alpha$

$\alpha$ = angle of attack

$C_{D_o}$ = drag at zero angle of attack

The angle of attack chosen for L/D up to M = 7.0 resulted in maximum values of L/D. However, above M = 7.0 values of L/D slightly lower than L/D (maximum) were used in order to operate at a higher angle of attack and therefore higher altitude and lower skin temperatures.

A plot of L/D versus Mach number for the glide flight is presented on page 45.

The re-entry path is considered to be a ballistic path with the glide configuration assuming zero angle of attack.

## PERFORMANCE SUMMARY

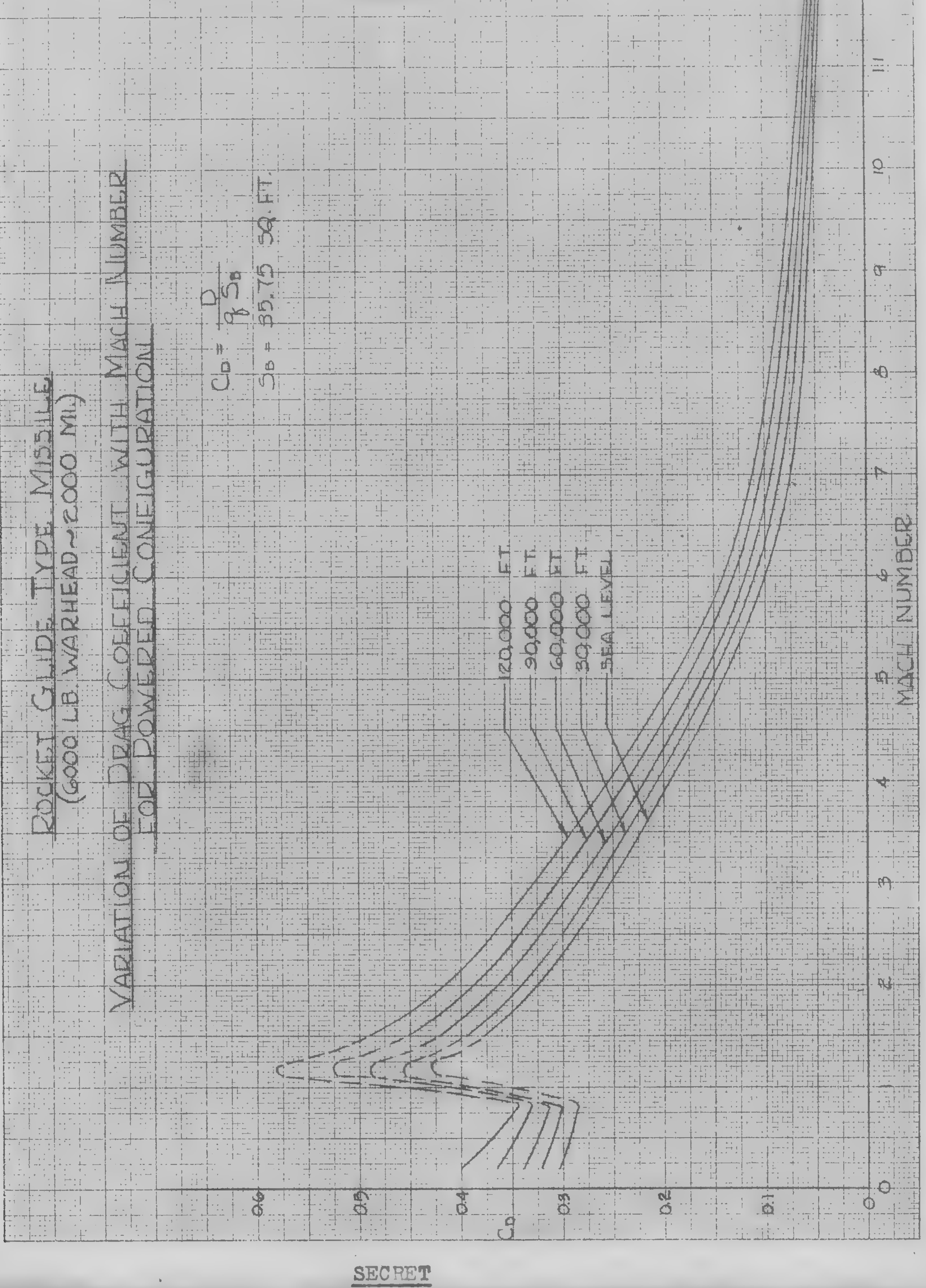
| | Booster Stage I | Final Stage II |
|---|---|---|
| Weight Full | 74,775 lbs | 17,492 lbs |
| Weight Empty | 20,201 lbs | 9,926 lbs |
| Weight of Fuel (total) | 54,574 lbs | 7,566 lbs |
| Thrust at S.L. | 108,000 lbs | 28,000 lbs |
| Specific Impulse at S.L. | 210 sec | 240 sec |
| Fuel Consumption | 514 #/sec | 133.3 #/sec |
| Mass Ratio | 3.695 | 1.762 |
| Initial Acceleration | 1.445 | 1.83 |
| Time of Burning | 106 sec | 568 sec |

| | Performance at End of Booster Stage | Performance at End of Final Stage |
|---|---|---|
| Velocity Along Flight Path | 7049 ft/sec | 11,290 ft/sec |
| $S_Z$ - Altitude | 87,000 ft | 115,000 ft |
| $S_X$ - Range | 190,000 ft | 690,000 ft |
| Flight Path Angle | 12.2° | 0 |
| Missile Attitude | 0 | 0 |

Performance at End of Glide

Altitude = 52,000 ft
Velocity = 2,000 ft/sec
Range = 2,005 miles

SECRET



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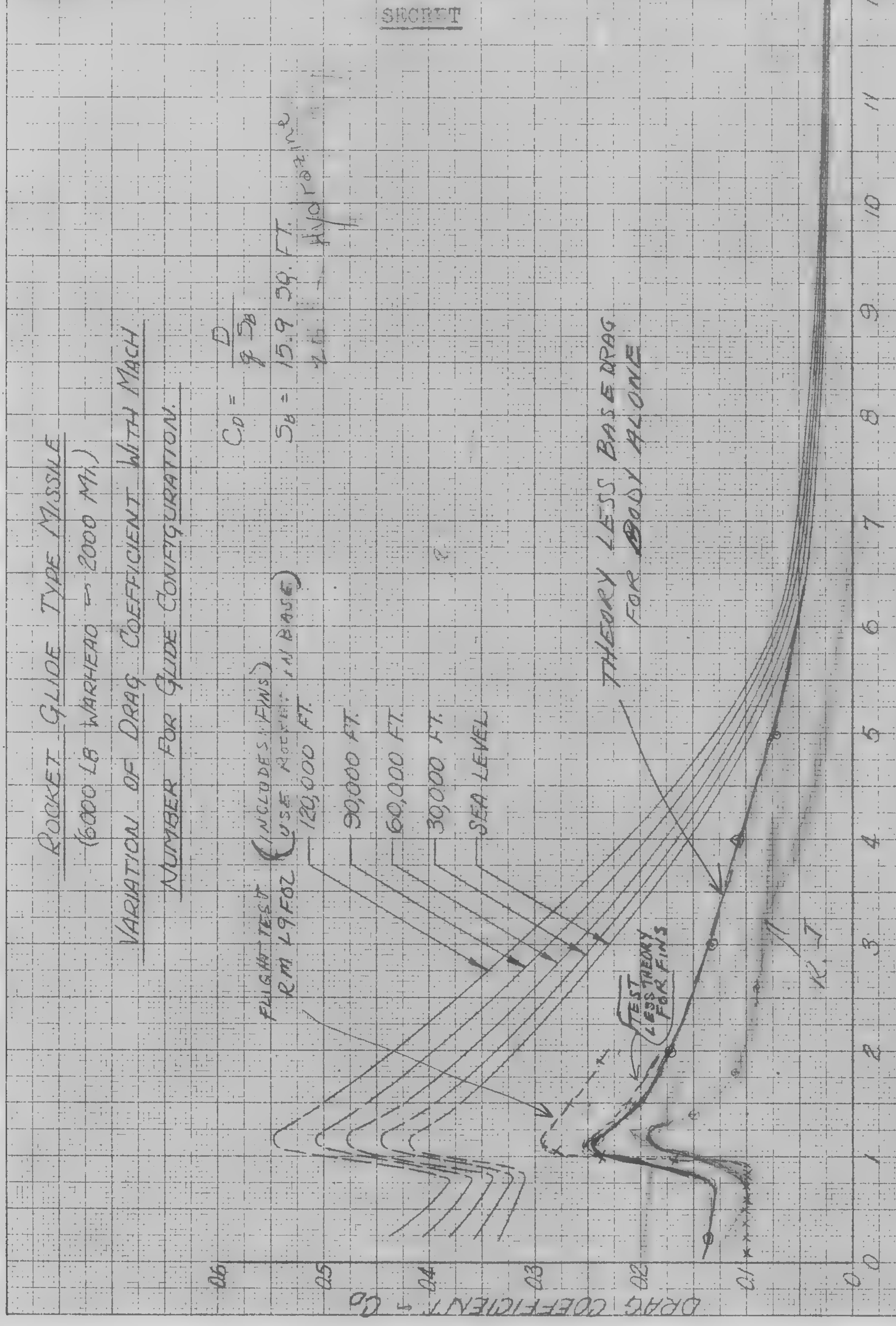
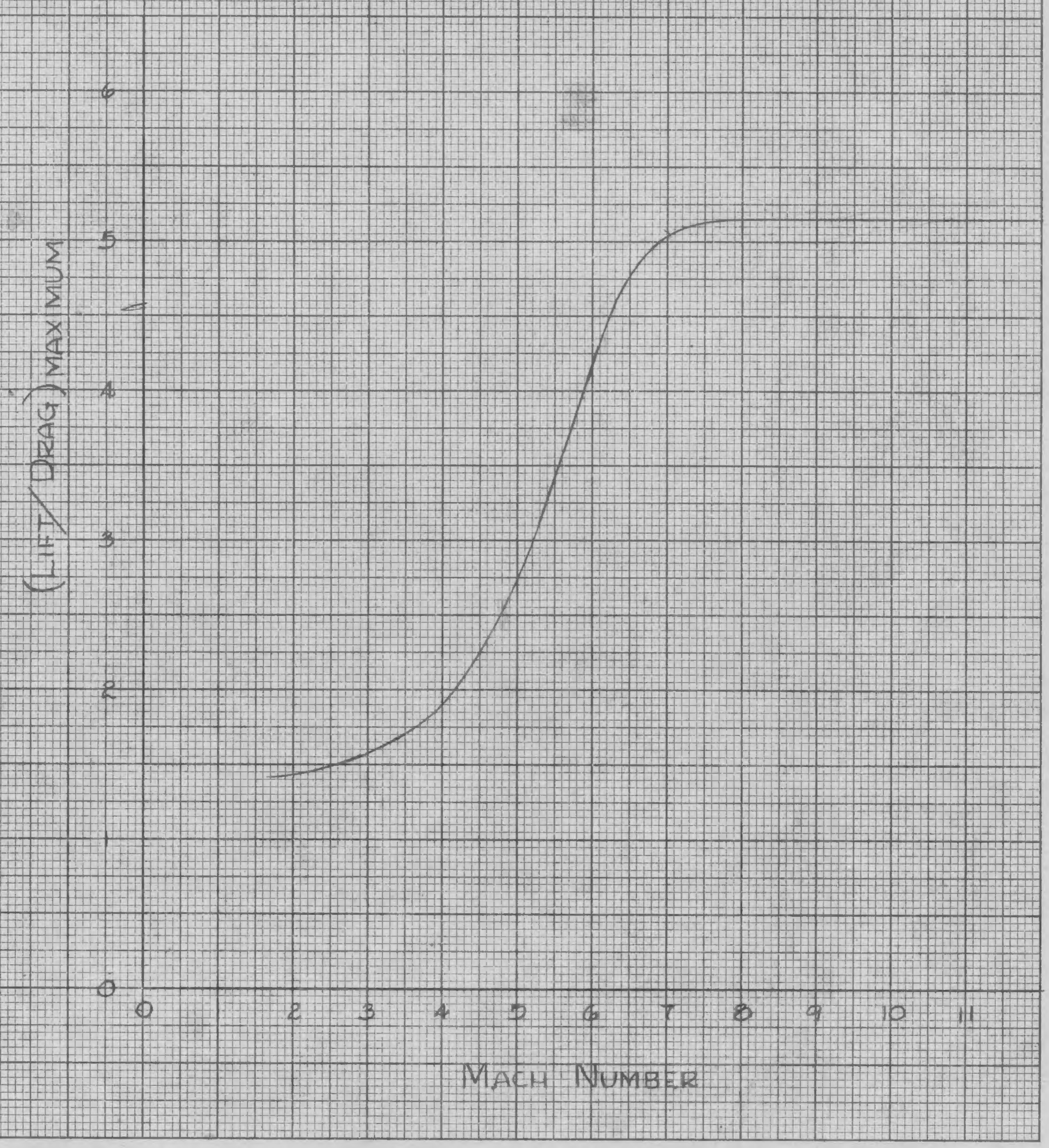

SECRET
ROCKET GLIDE TYPE MISSILE
(6000 LB WARHEAD – 2000 MI.)
VARIATION OF DRAG COEFFICIENT WITH MACH
NUMBER FOR GLIDE CONFIGURATION.
$C_D = \frac{D}{q S_B}$
$S_B$ = 15.9 SQ. FT.
FLIGHT TEST (INCLUDES FINS)
RM L9F02L (USE ROCKET IN BASE)
120,000 FT.
90,000 FT.
60,000 FT.
30,000 FT.
SEA LEVEL
THEORY LESS BASE DRAG
FOR BODY ALONE
TEST LESS THEORY FOR FINS
MACH NUMBER
DRAG COEFFICIENT → $C_D$

SECRET

# ROCKET GLIDE TYPE MISSILE

(6000 LB. WARHEAD - 2000 MI.)

## VARIATION OF LIFT TO DRAG RATIO WITH MACH NUMBER ALONG GLIDE PATH



F. Guidance: The methods of mid-course guidance applicable for a glide path trajectory were presented on page 24 of reference 1. In addition to guidance for the glide path the incorporation of a self contained system of angle of attack control is deemed desirable. Such a system would allow the programming of angle of attack throughout the flight thus permitting (1) flight at best lift to drag ratios at all times, or (2) predetermined control of the flight altitude within wide ranges.

The angle of attack programming signal can come from any one of three sources or combinations of these sources, (1) elapsed time, (2) altitude by positive altimiter, or (3) Mach meter.

It is possible that some disturbance might start the configuration on a skip path. If the angle of attack signal originates from either of the last two sources such a skip would be damped out.

G. COST ESTIMATE

The following Cost Estimates for the 6,000-pound warhead rocket missile are based on the same cost analysis presented on Page 2.

COST ESTIMATE

6,000 Pound Warhead Rocket Missile

| | | Ballistic Warhead 1,000 Mile Range | | Glide Warhead 2,000 Mile Range | |
|---|---|---|---|---|---|
| | Unit Cost | Wt. | Cost | Wt. | Cost |
| Airframe | $ 27.00 | 3,134 | $ 84,500 | 4,085 | $110,000 |
| Power Plant | 80.00 | 2,110 | 169,000 | 2,110 | 169,000 |
| Guidance | 100.00 | 300 | 30,000 | 440 | 44,000 |
| Warhead | ---- | 6,000 | ---- | 6,000 | ---- |
| Total Wt Empty | ---- | 11,544 | $283,500 | 12,635 | $323,000 |
| Oxygen | .04 | 33,832 | 1,310 | 33,832 | 1,310 |
| Alcohol | .10 | 27,065 | 2,707 | 27,065 | 2,707 |
| $H_2O_2$ | .16 | 1,243 | 200 | 1,243 | 200 |
| Total Fuel | ---- | 62,140 | $ 4,217 | 62,140 | $ 4,217 |
| Total | ---- | 73,684 | $287,717 | 74,775 | $327,217 |

## REFERENCES

1. CVAC Report No. ZP-48-35003, "Proposed Development Program on Rocket Type Missile".

2. CVAC Report No. ZA-6002-002, "Final Aerodynamics Report on MX-774 Single Stage Test Vehicle".

3. Rand Report No. RA-15065, Long-Range Surface-to-Surface Rocket Vehicles Preliminary Investigations and Results. - Aerodynamics

4. Rand Report No. RA-15063, Long-Range Surface-to-Surface Rocket Vehicles Preliminary Investigations and Rockets - Flight Mechanics.